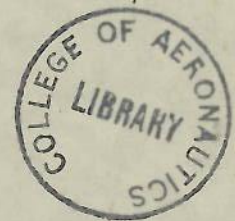


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THE COLLEGE OF AERONAUTICS
CRANFIELD



AIRCRAFT DESIGN STUDIES - VARIABLE SWEEPBACK
NAVAL AIRCRAFT

by

D. Howe



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Aircraft Design Studies - Variable Sweepback
Naval Aircraft.



by

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SUMMARY

A preliminary survey of the possible applications of variable sweepback indicated that a substantial improvement in performance can be expected when the concept is applied to a naval strike aircraft. In order to assess this performance gain and to obtain experience of the engineering problems involved, the subject of the design study by the students in the Department of Aircraft Design during the 1964 academic year was chosen to be a variable sweepback naval strike aircraft.

The aircraft has a maximum take off weight of 60,000 lb. and a limiting Mach number at altitude of 2.5. Various payloads up to a maximum of 4000 lb. can be carried externally over ranges which vary up to 4000 n. miles according to the role. Alternative mechanical arrangements for the wing hinge system were investigated.

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1.0 Introduction

The aerodynamic advantages of variable geometry in the form of changes in wing planform have been obvious for several decades. However it is only relatively recently that technology has advanced to the point where the full exploitation of this potential has become mechanically feasible. Even now the attendant penalties of increased weight and complexity are such that the application of variable sweep is economically worthwhile only on a small number of specific types of aircraft (1). Of these the supersonic airliner and strike aircraft deserve special mention. The systematic study of the problems of supersonic transports in the Department of Aircraft Design at the College of Aeronautics (2) naturally lead to some consideration of variable sweep in this connection (3), (4). However it soon becomes obvious that these investigations could not lead to realistic results unless a much more detailed knowledge of the mechanical aspects of the problem was obtained. It was therefore decided to carry out a project study of an aircraft which incorporated variable sweepback as a feature of design.

A preliminary survey indicated that the greatest performance gain resulting from the application of variable sweep would be realised in the case of a naval strike aircraft. This type was chosen for investigation by the students in the Department during the 1964 to 65 academic year. The list of those who participated in this study together with their individual responsibilities is given in Appendix A.

2.0 Specification of the S64 Naval Strike Project

It was necessary to propose a hypothetical performance specification in order to crystallise the general concept of a variable sweep naval strike aircraft into a specific project design. This specification was deliberately chosen to enable the full advantages of variable sweep to be realised and it was, of necessity, somewhat arbitrary from the real operational aspect.

The aircraft is intended for operation off aircraft carriers in three basic roles.

- (1) Strike operations against land or sea targets. A maximum payload of 4000 lb is carried externally over a design radius of operation of up to 750 n miles. The maximum low level design Mach number is 1.4.
- (2) Search operations, the maximum endurance being of the order of the order of eight hours whilst flying at a Mach number of 0.75 at altitude.
- (3) Interception operations carrying four air to air guided missiles. A normal flight Mach number of 2.0 is supplemented by a dash to M = 2.5 during interception.

The search and strike roles implied the need for two crew members, and side by side seating was considered to be mandatory to ensure the maximum operational efficiency.

The limitations imposed by the aircraft carrier were established partly by a consideration of known features of Royal Navy vessels and

partly by assumptions as to what might be expected of a ship used as a base for such an aircraft. The maximum take off gross weight was limited to 60,000 lb. and the maximum folded length, width, and height to 57.3 ft., 30.0 ft., and 18.0 ft. respectively. Operation off a 150 ft. long B.S.6 steam catapult was stipulated with nosewheel tow launch. A 195 ft. long water spray arrester gear was assumed to be available.

An aircraft life of not less than 3000 hours, equivalent to 5000 flights, was specified in conjunction with a design normal acceleration factor of 8. The proposed airframe flight limitations are shown in Figure 6.

Two independent power plants were stipulated.

3.0 Overall Configuration of the Design

A general arrangement drawing of the project aircraft is shown in Figure 1, whilst Figures 2 and 3 are photographs of models.

The wing has a low mid setting on the rectangular cross section fuselage. The fixed inboard panels have a cropped delta shape with a leading edge sweepback of 66° which coincides with that of the outer wing in the high speed flight position. The outer wing panels are almost rectangular in shape, having very little taper. The trailing edges have a circular arc shape near to the root, the centre of the arc coinciding with the wing pivot position. In the lowspeed position the leading edge sweepback on the outer panel is 13° and the overall aspect ratio is just under five. The high speed position has an aspect ratio of approximately two, the gross wing area being 4.1% less than the 723 sq. ft. of the low speed configuration. The straight portions of the trailing edge are occupied by what are effectively full span double slotted flaps.

The tailplane is delta shaped and each of the two panels is capable of independent movement. The fin is also delta shaped and it is fitted with a conventional rudder. It is supplemented by a ventral fin of significant size which also serves as a tail bumper.

Two bypass turbojet engines are located side by side in the centre fuselage immediately aft of the wing. The engine air intakes are unusual in that they are positioned over the top surface of the fuselage, just aft of the cockpit region. This location was chosen in preference to fuselage side intakes as it is considered that it affords a considerable relief to the serious intake ingestion problem of naval aircraft. It also enables a neat two dimensional variable wedge system to be designed, this type of intake having been adopted for a previous design study of a Mach 2.2 airliner (2). It is, nevertheless, fully realised that the upper fuselage position may be ruled out by consideration of adverse flow over the nose and canopy, especially at high angles of attack. The reheat system shown in the drawings, for example the internal layout, Figure 4, also utilises a two dimensional variable wedge arrangement. This suits the basic fuselage configuration rather than the annular nozzle systems which are now favoured.

Claim shell air brakes are located at the base of the fin, above

the reheat nozzles. The fuselage cross section shape was considerably influenced by area ruling consideration. These were primarily directed towards the achievement of low wave drag in the low level supersonic dash role. In this context the air brake played a substantial part in smoothing the tail of the area distribution.

The main units of the conventional tricycle undercarriage carry twin, side by side, wheels and are cantilevered off the lower outer fuselage edges. They retract rearwards into separate bays positioned below the powerplants. These bays also house the wing trailing edge regions when the wing is swept back. The nosewheel is heavily loaded by the nosewheel tow catapult requirement, and the layout is unusual. The drag strut and the towing attachment connect to the axle. The torque links are special in that they can be locked to react the high leg compression loads during catapulting. The unit retracts rearwards. A conventional sting deck hook is attached to the rear fuselage immediately aft of the mainwheel bays.

The main equipment bay is located around the nosewheel bay, immediately aft of the cockpit. Access to this region is through the nosewheel bay sides. The powerplants are installed through doors located in the top of the fuselage with subsidiary access for servicing through the roofs of the mainwheel bays. The remainder of the fuselage is used as integral fuel tanks, including the volume below the jet pipes. Fuel is also carried in the centre wing box and the forward regions of the inner wing, but there is no fuel in the outer wing panels.

One of the major layout problems encountered in the design was the achievement of the required folded dimensions. This was especially difficult on the length, which inevitably tends to be relatively great on a supersonic design in order to restrict wave drag to an acceptable level. The flight length of just under 75 ft. is reduced to 57.2 ft. for stowage by folding the nose forward of the cockpit and opening the air-brakes to a lateral position. The requirement for a folded span of 30 ft. necessitated the complication of a further degree of sweepback beyond the high speed flight position, whilst the height restriction resulted in the adoption of a fin of lower aspect ratio than was otherwise desirable. The location of the external weapons for typical operational roles is shown in Figure 5. The complication of store positions on the other wings has been avoided.

4.0 Control and Performance

Control of the aircraft is unorthodox in that there are no ailerons. Since the choice of pivot position for the wing has the effect of precluding the use of the inboard trailing edge for flaps it is necessary to extend the flaps out to the tip. The basic method of roll control is the differential movement of the two independent tailplane panels. Adequate roll power is achieved by this means in most flight regimes although in certain combined rolling and symmetric manoeuvres the tailplane loads are large. However the precise roll control re-

quired for landing on an aircraft carrier cannot be met by differential movement of the tailplane halves. Wing spoilers were considered as an obvious possibility for overcoming this difficulty, but they were not favoured because of the implied loss of lift which could be critical in these exacting circumstances. The alternative which was adopted was a differential movement of the flaps about the normal fully extended setting. The full effect of this proposal has not been fully investigated, and in particular the adverse yawing motion may require to be corrected through an interconnection with the rudder. It has been estimated that the flap setting changes would only need to be approximately $\pm 5^\circ$ when the technique is used in conjunction with the differential tailplane.

Longitudinal control and trim is obtained by symmetric movement of the two parts of the tailplane, no auxiliary surfaces being provided. The roll control system prevents the use of a mechanical interconnection between the separate parts of the tailplane, but this is made good by the stiffness of the control system.

Directional control is derived from the conventional rudder.

It is not envisaged that the variation of wing sweepback from 13° to 66° should be continuously available throughout the speed range. Basically there are just the two extreme positions, the transition between them being undertaken at subsonic speeds in the range of Mach number between 0.75 and 0.85. Strike and interception operations are carried out with the wings fully swept, but search or loiter flying uses the lowspeed position. Whilst this restriction introduces some aerodynamic performance penalties it is structurally advantageous in that the full manoeuvre load factor need not be applied at intermediate positions. The design load factor during wing transition was determined by gust considerations to be approximately 50% of the maximum value. The trim change as the wing is swept is primarily a function of the body effect, as may be seen by reference to Figure 7. The total change in aerodynamic centre position over the flight speed range is approximately 2 ft. for the low altitude supersonic dash case, and rather more than 3 ft. for the high altitude, maximum Mach number case. The allowable centre of gravity range has been determined with reference to the movement of the aerodynamic centre.

The estimated zero lift and induced drag coefficients for a complete range of flight conditions are shown in Figures 8 and 9 respectively. Barrett (5) has used these values in conjunction with assumed powerplant characteristics to evaluate the overall performance using an energy height technique. This investigation was carried out in conjunction with the design of the fuel system and it was used to evaluate the rates of fuel usage. The results are summarised in Figures 10 to 12. The first of these diagrams shows the calculated drag force at 10,000 ft. altitude as a function of Mach number for the design weight of 60,000 lb. The engine performance in terms of thrust and specific fuel consumption with the reheat on is also shown. Figure 11 shows the lines of rate of change of energy height for the design weight together with a probable climb path and the overall flight limitations. On the basis of this type of diagram it has been estimated that the aircraft can climb from sea

level at 140 knots to 40,000 ft. at Mach 2.5 in a minimum time of 8 minutes, or alternatively use a minimum quantity of 7000 lb. of fuel. This fuel consumed is reduced if the reheat is not used during the initial phases of the climb. The absolute ceiling is 55,000 ft. at 60,000 lb. weight, and about 62,000 ft. at 45,000 lb. weight. Typical range performance for different operating conditions is shown in Figure 12. The long range cruise at a Mach number of 0.75 and the low level flight at a Mach number 0.9. have been estimated assuming that the reheat will not be used.

The estimated catapult end speed of 118 knots implied a take off lift coefficient of 1.5 when there is a 20 knot head wind. The actual take off lift coefficient when the nosewheel is in contact with the deck is estimated to be 1.35 so that it would be necessary for an attitude indicator or control to be employed for launches at the design weight. The maximum acceleration is 4.2g. When landing at 50,000 lb. weight the terminal approach speed has been calculated to be 112 knots. This implies a deceleration of 4g with the 195 ft. water spray arrester gear. Evans (6) and Craig (7) used digital and analogue computers, respectively, to study the actual motion of the aircraft during arrested landing with a view to ascertaining the undercarriage and deck hook parameters.

5.0 Hinge System and Fairing

The position of the main wing hinge was determined after consideration of several conflicting requirements. From the point of view of stability and control it is desirable to minimise the change in the fore and aft position of the aerodynamic centre both with variation of sweep-back and Mach number. The best hinge location to meet this requirement was either too far aft on the chord or too far out along the span to have adequate structural depth. It was also found to complicate the problems associated with the adequate fairing of the wing in the basic sweep configurations. In fact the fairing was found to be the major issue and the solution which was adopted ultimately yielded an acceptable compromise of the structural and aerodynamic requirements.

Figure 13 shows the basic hinge geometry. From this drawing it can be seen that relative to the high speed configuration the hinge is located at approximately the mid chord, 34% along the semi-span. The position on the low speed wing is 21% of the semi-span. The fairing of the wing is illustrated in Figure 14 which shows the wing in three sweep positions. A flow control fence is positioned along the forward half of the chord, some 3.2 ft. outboard of the hinge, and this is used to mask the change in depth required to house the leading edge beak in the lowspeed configuration. The trailing edge inboard of this chord is based upon the use of cylindrical sections which are housed within the fuselage when the high speed configuration is in use. The outer wing panel has a spanwise step which is aligned with the flow control fence when the wing is unswept. There is a corresponding step on the aft edge of the fixed inner panel so that when the wing is fully swept the two butt together to give a smooth contour. The outer panel step lies across the flow direction during the process of sweeping the wing, but since this occurs over a period of only 20 seconds the drag penalties are of no consequence. Working clearances are required on the fence

and fuselage side to cater for structural distortions and these are closed by inflatable seals. Stowage with the minimum folded span necessitates the drooping of the trailing edge flaps so that a further portion of the wing can enter the fuselage.

The aerofoil sections are dictated to some extent by the fairing configuration, especially at the inner trailing edge. As can be seen by reference to Figure 15 the outer sections are based upon the use of a biconvex aerofoil with a modified, rounded, leading edge. Nearer to the root the maximum section depth tends to move forward on the chord. The thickness chord ratio at the fence is approximately 12% in the low speed configuration and 6% in the high speed case.

Two mechanical designs for the hinge system have been investigated in some detail. The first of these uses a single large pin, the structural bending loads being reacted as horizontal shears, whilst the second is a hybrid shear arrangement with a pin and track.

The horizontal shear design is shown in Figure 16. Separate ball screw actuators driven by hydraulic motors are used to move the outer wing panels. They are positioned between the fuselage side and the front spar of the outer wing, and are connected by a cross shaft. This is used for synchronisation and it enables either actuator to drive both wings in an emergency. A lock is provided at the front spar root for the low speed position and there is also a support at the rear spar which engages in the high speed position. The hinge itself is carried at the extremity of 3.5 ft. wide single cell centre wing box.

The alternative pin and track scheme is illustrated in Figure 17. The particular arrangement shown was arrived at after consideration of several other track locations. The tracks are positioned forward of the hinge in the fixed inner wing, and penetrate into the fuselage side as well as the wing box. However the latter is some 2 ft. wider than that used in the single pin design. The forward halves of the outer wing panel structures extended inwards and each carries three track pads. Three, rather than two, pads are used to enable track bending loads to be reduced in the critical design cases. The actuators also pass through the centre wing box and hence are not readily accessible. Small doors are necessary in the outer wing leading edge beaks to enable them to clear the track extensions when they are housed within the inner wing.

It is apparent that the second scheme is more complex and less readily accessible for maintenance than the first arrangement. Nevertheless a weight comparison indicates that it could well have a smaller weight penalty. The weights estimated for the two schemes are 5140 lb. for the single pin and 5070 lb. for the hybrid design. These include the centre wing box structure, hinge and outer wing lugs but not the actuators.

It would appear that a reconsideration of the basic hinge geometry and fairing would eliminate some of the difficulties associated

with the pin and track design, in which case it would be preferable. With-in the geometry specified the single pin scheme is more suitable and the design of the aircraft was based upon its use.

6.0 Description of Design Details

A key diagram of the complete airframe giving the location of the main structural members is shown in Figure 18. Where possible low creep light alloys have been used for structural components. Extensive application of titanium alloys is necessary in the rear fuselage and high strength steels are employed in high stress regions such as the hinge and centre wing box, undercarriage, and tailplane mounting frames.

6.1 Wing Structure - Single Pin Scheme

The single pin hinge arrangement allows a relatively simple structural layout to be adopted. The hinges are carried at the extremities of the single cell inner wing box which is continuous across the fuselage. The outer wing panels employ a three cell box structure with a minimum number of ribs.

The design loading cases for the wing in the two extremes of sweep-back both occur in 8g manoeuvres whilst the aircraft is flying at sea level and with the maximum take off weight located at the forward centre of gravity limit. The maximum factored vertical shear force at the hinge is 186,000 lbf. which arises with the wings forward and a Mach Number of 0.85. Flight at a Mach Number of 1.51 with the wing swept back gives the maximum factored bending moment of 2,350,000 lbf.ft. The corresponding maximum loads on the inner wing are a centreline factored bending moment of 2,650,000 lbf.ft. and factored shear forces of 272,000 lbf. at the fuselage side.

6.1.1 Inner Wing Structure

The inner wing structure is designed to be integral with the centre fuselage, the main structural box being used as a spine onto which the frames and ribs are assembled. The box is 43 inches wide and the front and rear spar webs are coincident with, and form part of, the fuselage frames which are placed at stations 36.65 ft. and 40.25 ft. respectively. All the box structure is fabricated in high strength steel and considerable use is made of machined forgings. The individual components are mainly joined together with high shear pins and bolts but welding is used in certain regions. Figure 19 shows the basic construction.

The forged D6AC skins of the box have a basic thickness

of 0.16 inches on the upper and 0.13 inches on the lower surface. Both faces are supported by spanwise integrally machined webs of 2.0 inches depth and 0.2 inches width which are located at 3.6 inches pitch across the chord. The skins terminate in the integrally forged female lug fittings. The lugs have a radius of 12.4 inches and are 1.10 inches thick.

There are five single piece S99 integrally machined ribs. These are located on the aircraft centreline, at the fuselage sides, and intermediately to give a pitch of approximately 26 inches. The two intermediate ribs are of channel section with a 0.15 inches thick web which is supported by vertical stiffeners. The other ribs are of I section and have booms which are 2.8 inches wide and 0.3 inches thick, but which are otherwise similar to the intermediate ones.

The front and rear webs of the box are portions of the two fuselage frames and carry angles for the attachment of the top and bottom covers. Outboard of the fuselage sides they are extended by D6AC machined forgings which connect to the ends of a 0.2 inches thick D6AC curved web. These extensions taper in thickness from 0.3 inches outboard to 0.13 inches at the fuselage sides and they are stiffened by vertical legs. Each of the curved webs is further attached to three similar, short webs which join it to the skins and fuselage side ribs. All of these vertical members are welded to the skins.

A single vertical pin engages with a corresponding lug on the front spar of the outer wing when it is in the unswept position and a similar arrangement is used at the rear to lock the wing in the fully swept flight position. The wing sweep actuators are placed between the front spar roots of the outer wings and locations on the sides of a fuselage frame.

The main undercarriage units are attached to the rear spar bulkhead at the fuselage side and drag loads are distributed into the main structure by means of fore and aft members which are placed within the wing box.

The remainder of the centre wing is constructed in D.T.D. 5070A light alloy. The four subsidiary spars are extensions of alternate fuselage frames and there are five intercostal ribs on each side of the aircraft. Both the spars and ribs terminate at a swept web to which the small leading edge riblets are attached. The skins over the region of the hinge are sandwich panels supported by riblets mounted off the primary structure and the airflow fence. These skin panels have 18g thick D.T.D. 5070A faceplates which are separated by 0.7 inches deep honeycomb cores.

6.1.2 Outer Wing Structure

The outer wing is not used as a fuel tank. It is built up from a relatively small number of components, and the four spars are

located at 17%, 33%, 47% and 65% of the chord. The rearmost of these supports are full span flaps shrouds and tracks. Like the skins the spar webs are machined from D. T. D. 5084 forgings. At their inner ends the skin and spars are joined to the FV520 male hinge lugs as is shown in Figure 20. The lugs themselves are formed by pairs of machined plates having a total thickness of 2.0 inches. Welding is used to join them together and to spars and a complex of shear webs. The lug to skin joint is in the form of a splice, the substantial kink loads which result from this being reacted in torsion in a chordwise box formed by two ribs which are placed 20 inches apart.

The skin thickness varies from a maximum value of one inch at the root of the top surface to approximately 0.1 inches at the lower surface tip. The inner portions of the skins are not supported by spanwise members other than the spars but outboard where the thickness falls below 0.45 inches, webs of 0.75 inches depth and 0.35 inches width are introduced. They are placed at 3.75 inches pitch across the chord. Intermediate between the root and the tip there are three main ribs. These have a pitch of eight feet and coincide with the flap support brackets. They are intercostal between the spars and are machined from L65 forgings.

The 18g leading edge skin is stiffened by 1.5 inches wide by 0.75 inches deep spanwise corrugations. It is supported by 20g pressed ribs located at approximately 32 inches pitch.

6.1.3 Hinge Assembly

A section through the hinge assembly is shown in Figure 21. The hinge pin is actually two separate, cylindrical, shallow members which mate with the reinforced PTFE 'Fibreslip' radial shell bearings. The bearing has a small taper angle and this is used in conjunction with a ring nut and thrust washer of the same material to adjust the assembly. The radial and thrust bearing surfaces are carried on housing members which are keyed to the male, outer wing lugs. The ring nuts connect the hardened and polished steel pins to the female lugs.

The radial bearings have a mean diameter of 14.2 inches and have a nominal vertical spacing of 12.6 inches. The maximum design static bearing pressure is 70,000 p.s.i., and the working design pressure of 23,500 p.s.i. occurs when the aircraft encounters a 4g gust condition during wing sweep. The bearing rubbing velocity is approximately 1.5 ft. per minute.

6.1.4 Trailing Edge Flaps

The double slotted flaps occupy the whole span of the

outer wing although they are adjacent to the fuselage side only when the wing is in the fully swept flight position. The flaps are drooped for stowage on the carrier. Each of the flaps is some 19.4 feet long and is carried on three sets of flap tracks. The conventional construction employs a single web with pressed channel section ribs to support the 18g thick D. T. D. 5070A skins.

6.2 Wing Structure - Pin and Track Scheme

The use of a pin and track arrangement to support the outer wing panels results in a structural configuration which is more complex than that of the alternative scheme discussed above. As designed the outer wing is supported by bearing pads which move along a track mounted on the inner wing.

6.2.1 Inner Wing Structure

The main inner wing structure is shown in Figure 22. The two cell box has a total width of almost 5.5 feet which is some two feet more than that employed for the single pin scheme. However the plan-form geometry of the track is such that it makes a considerable excursion into the front cell of the box and the hinge is effectively carried only on the 1.5 feet wide rear cell. Many of the components are machined from D6AC steel forgings.

The skins over the aft cell extend across the aircraft and incorporate the female lugs for the hinge. They are forged in D6AC and have basic thicknesses of 0.25 inches over the top and 0.4 inches over the bottom of the box. At the outer extremities the thickness is increased to 4.15 inches from which the one inch thick female lug ends are machined. These are of 5 inches radius. Spanwise stiffeners of 2.0 inches depth and 0.25 inches width are placed at 4.0 inches pitch on the inner surfaces of both skins.

The front cell skins are 0.20 inches thick D6AC plates and they have 2.0 inches deep by 0.25 inches thick zed section stringers placed at 3.5 inches pitch. These stringers are attached by $\frac{1}{4}$ inch diameter bolts placed at 1.5 inches pitch. The only complete rib is located on the aircraft centreline. This is an I section D6AC machined component with a basic web thickness of 0.25 inches and booms of 2.0 inches width. Ribs are also located at the fuselage sides and intermediately between these and the centreline. The latter pair of ribs coincides with the inner ends of the tracks which actually form short lengths of the ribs just forward of the central spar web. The track is also forged and machined in D6AC and it effectively acts as a curved web on the outer face of the forward cell. The basic thickness is 1.14 inches and there are 0.25 inches thick skin attachment flanges.

The front and rear webs of the box form part of the relevant fuselage frames which are similar to those employed for the previous scheme. The centre web and extensions of the front and

rear webs outboard of the fuselage are D6AC plates welded and bolted to the skins. The wing operating actuators pass through the rear cell, between the track and fuselage side ribs. They are attached to the aft face of the rear web at the track web stations.

6.2.2 Outer Wing Structure

The outer wing construction is also based on a two cell box the inner end of which is shown in Figure 23. The hinge is located on the aft face of the centre spar and the hinge bracket is connected to the rear spar by means of an inclined rib. Each hinge fitting is a D6AC forging which is bolted to the inclined rib, centre spar web, and a front cell rib. Three track pad mountings are carried on the end of the front cell which extends some 3.5 feet inboard of the hinge.

The D6AC skins of the front cell inboard extension are 0.2 inches thick and have 2.0 inches deep by 0.35 inches wide integrally machined stiffeners at 3.3 inches pitch across the chord. All three spars and the inclined hinge support rib are built up from machined components in D6AC. The actuator attaches to the aft face of the centre spar, a separate lug fitting being bolted onto the spar web for this purpose.

The design of the structure out board of the hinge has not been investigated in detail.

6.2.3 Hinge and Track Bearing Assemblies

The details of the hinge and the track bearings are also illustrated in Figure 23.

The hinge design is somewhat similar to that shown for the single pin scheme of Figure 21, but it is much smaller as the bearing diameter is only 6.0 inches. The bearing rings, or shear pins, are located in place by a full depth, 3.0 inches diameter, tension pin. The estimated maximum static and working bearing pressures are 60,000 p.s.i. and 20,000 p.s.i. respectively.

The track bearing assembly consists of a triangular shaped fitting which is attached to the outer wing box extension by a single pin of 2.0 inches diameter. This pin is mounted in a 4.2 inches diameter eccentric bush so that vertical adjustment can be made. Each track pad assembly carries three bearing pads. Two of the pads bear on the lower track surface and react the upward wing bending loads which correspond to positive normal manoeuvres, whilst the third bears on the top of the track and thus reacts negative manoeuvre loads. Each of the pads is 3.0 inches long by 2.4 inches wide and is mounted in a hemispherical bush to cater for track misalignment and structural distortion. Both the bearing pad and the hemispherical bush use

reinforced PTFE 'Fibreslip' bearing material.

The stress distribution in large bearings of the type associated with variable sweep wing hinges and track pads was investigated separately by Noffsinger (8). Both theoretical and photoelastic methods were used in an attempt to assess the relationship between the maximum and mean bearing stresses and the influence of the bearing and lug geometry.

6.3 Fuselage

The form of construction and the materials used vary considerably along the length of the fuselage. Light alloys, titanium and steel alloys all occur in individual components. The structure is complex and in addition to analysis by simple methods in conjunction with appropriate assumptions, a serious attempt was made by Bright (9), Clark (10) and Northover (11) to apply force - displacement techniques. Lack of adequate computer storage facility somewhat restricted the use of these solutions.

The design vertical shear force and bending moments occur when the aircraft is manoeuvred with a normal factor of eight at a Mach number of 2.5. In these circumstances the ultimate shear force at the front spar station is 335,000 lbf. and the corresponding maximum bending moment is 3,700,000 lbf.ft.

6.3.1 Nose Fuselage Structure

The portion of the fuselage forward of the air intakes is designed in light alloy to specification D. T. D. 5070A. The 2.5 inches deep pressed channel section frames are located at pitches which vary in the range of 7 to 12 inches. Their inner flanges are locally reinforced by the addition of riveted angles. A basic thickness of 0.05 inches is used for the outer skin, but this is reduced to 0.035 inches by chemical etching except at the attachments for the frames and stringers. These latter are of inverted top hat section and are placed at a mean pitch of 8.0 inches around the periphery of the section.

The cockpit region is designed to cater for an operating pressure differential of 4 p.s.i. It is closed at the forward end by a stiffened, non-circular, domed bulkhead. The crew ejector seats are attached to the flat, rear, sloping pressure bulkhead. The 10g thick web of this rear bulkhead is reinforced by some fifteen vertical top hat section stiffeners. An L65 forging is located along the lower edge of the member and provides for the attachment of nose undercarriage leg trunnion fittings. A 12g thick channel section edge member connects the tops of the frames along the boundary of the canopy cutout.

The nosewheel bay extends aft from the sloping bulkhead and is closed at the sides by a pair of vertical shear webs which

run aft to the rear of the bay. The edge members are extruded angles. The bay is completed above by a flat roof which runs across the width of the fuselage and serves as the bottom surface for the forward fuselage integral fuel tanks. It is of light alloy honeycomb sandwich construction.

The nose of the aircraft in front of the cabin houses the search radar scanner. It is arranged to fold sideways and rearwards so that the overall length may be reduced for ship board operations. Toggle fasteners are employed to lock it in the flight position.

6.3.2 Centre Fuselage Structure

For the purpose of description the centre fuselage is regarded as that part of the structure which is bounded at the forward end by the rear nosewheel bay bulkhead and at the aft end by the rear spar bulkhead. However the forward and centre fuselage have been designed to be built as one unit. The presence of the air intakes and ducts considerably complicates the structural layout. Forward of the wing box the construction is conventional in that it is similar to the nose fuselage, but there are differences in detail due to the heavier loads and intake cutouts.

The skin thickness varies between 0.057 inches and 0.064 inches and the 16g zed section stringers are placed at a pitch which varies from 5.0 inches along the sides to 3.5 inches at the top and bottom of the section. They are supported by the frames which are placed at a pitch of 9 inches. The wing box passes through the lower part of the section and is built integrally with the fuselage, the front and rear spars being part of the corresponding fuselage bulkheads. Over the region between the wing spars the skin thickness is increased to 0.1 inches but the frame spacing is opened out to 14 inches.

The cross section of the centre fuselage is divided by a vertical shear web located on the centreline between the intakes. The section forward of the wing box is further divided by a horizontal shear web which is located below the intakes. These vertical and horizontal webs form the boundary walls and roofs for the centre fuselage integral fuel tanks. They are fabricated as sandwich panels using a light alloy honeycomb core. The nosewheel bay rear bulkhead and part of the front spar bulkhead are tank end closures, and they also employ a honeycomb core sandwich construction. The vertical fore and aft shear web is attached at the lower edge to a box section keel member which runs aft into the rear fuselage. This is built up from 0.25 inches thick D. T. D. 5084 extrusions and is 10 inches wide by 7 inches deep. The nosewheel drag strut is attached to the forward end of the keel member and the deck hook to the aft end. It is thus used to distribute into the fuselage all the heavy fore and aft loads associated with catapult take off and arrested landing.

The rear spar frame is machined from D. T. 5084 forgings. It has a 0.2 inches thick web which is supported by 1.7

inches deep by 0.3 inches thick vertical stiffeners placed at 8 inches pitch across its width. Flanges are machined onto the web for the attachment of the vertical web and wing skins. The main undercarriage pintle brackets are bolted to the lower outer corners of the frame and are connected to forgings which run forward into the wing box to diffuse the drag loads into the structure.

6.3.3 Rear Fuselage Structure

The rear fuselage construction differs from that of the forward and centre fuselage in that it is almost entirely based on the use of titanium alloys, mostly of the 6Al, 4Va type. The two engines are installed in bays which are located immediately aft of the wing rear spar bulkhead. The main undercarriage units are housed below the engines in separate bays which also provide stowage for the wing trailing edge structures.

The frame pitch varies from 12 inches just aft of the rear spar to 18 inches at the tail end. Channel section pressings in 18g titanium D.T.D. 5163 are used except for the main attachment frames and tank bulkheads. The outer shell is built up from 20g titanium skins to which are spot welded 23g by 0.6 inches deep longitudinal corrugations. The engines are installed through large cutouts in the top of the fuselage and subsidiary access for servicing is from below, through the roof of the undercarriage bay. Extruded steel channel section edge members are provided at the outside edges of the engine bay cutouts and there are titanium tee section extrusions along the edges of the undercarriage bay. The two engine bay doors are load carrying and each one is attached by 28 quick release shear fasteners which clamp into conical housings. These are illustrated in Figure 24.

The central vertical web continues aft from the centre fuselage and in the region of the engine bays acts as a firewall. Like the floor between the engine and undercarriage bays in the rear fuselage, the vertical web consists of a titanium skin supported by longitudinal corrugations. The floor also extends some distance aft from the undercarriage bays and acts as a roof for the rear fuselage integral fuel tanks, these being located below the jet pipes. However the box section keel member terminates at the aft end of the undercarriage bays in the deck hook attachment fitting. The hook is a sting type with a centralising spring unit on the lateral pivot and a combined retraction jack-hydraulic damper across the vertical pivot. The hook itself is a steel investment casting and it is bolted to the end of the 4 inches diameter by 0.27 inches thick T45 tube. In the retracted position the hook is housed in a tunnel in the lower surface of the fuselage which is fabricated from a 0.18 inches thick D.T.D. 5084 extrusion.

Figure 24 shows the lower bulkhead at the aft end of the undercarriage bay. Machined from a D.T.D. 5084 forging this component incorporates a large removable panel for access to the integral tanks.

The tailplane assemblies are mounted on spigots which extend out from the sides of the fuselage. They are supported between a pair of substantial channel section frames which are placed 6.5 inches apart. These are machined from D.T.D. 5084 forgings but the 6.0 inches diameter spigots are S99 forgings. The two frames forward of the tailplane mounting and the next one aft coincide with the three fin spars. The centre one of these three is also used for the attachment of the tailplane actuators. Alternative designs for the fin spar frames were considered by Read (12). One possibility was the attachment of the fin to the top of the fuselage only, but this was discarded in favour of continuation of the spars down through the full depth of the section. As designed the fin frames are built up from steel plate webs and flanges, the front two being of box section and the rear one a channel. The rear end of the fuselage terminates in a frame to which the variable area exhaust nozzles are attached.

Clam shell air brakes are located at the base of the fin. These extend to a sideways position for stowage on an aircraft carrier. The brakes themselves and the operating mechanism are illustrated in Figure 25. They are fabricated from FV520 steel sandwich panels which have 30g skins and a 0.5 inches deep brazed honeycomb core. The shell is supported by pressed 24g, FV520 frames which are placed at 11 inches pitch. Two jacks are used to open the brakes. They are mounted on a forged S80 crosshead which moves along an S99 track. Links connect the air brakes to the crosshead and to the main structure.

6.3.4 Cockpit and Windscreens

The two crew members are seated side by side in the cockpit. In order to minimise the frontal area and at the same time to provide improved forward view and adequate ejection clearances the seats are inclined at an angle of 6° in plan, the front ends being closest to the centreline. Standard Martin-Baker rocket type ejector seats are provided. A special study of the temperature environment of the cockpit and the insulation and cooling requirements was carried out by Coles (13). The cockpit is supplied with air tapped off the low pressure end of the engine compressor and cooled to 10°C by a Freon refrigerator system. One inch thick Thermoflex insulation is placed on the inside of the fuselage skins.

The front windscreen is double vee shaped. Some details of the construction of this and the canopy are shown in Figure 26. Each windscreen panel has three separate layers of glass. The outer layer is primarily a heat shield and consists of a 0.4 inch thick alumino-silicate glass. There is an air gap of 0.5 inches width between this and the other panels which consist of two 0.47 inches thick glasses separated by a 0.4 inches vinyl interlayer. The pressure loads and bird impact are reacted in this rear panel assembly which is coated on the inside with Sierracin 90.

The two side portions of the canopy are hinged to a fore

and aft member located along the top centreline, and they open inwards for crew entry and egress. Each side canopy is formed from 0.5 inches deep aluminium honeycomb sandwich which has 25g thick D.T.D. 5070A facings. It is mounted on a D.T.D. 298 cast light alloy frame. The small window panels are similar in design to those of the main windscreen, but the thicknesses are reduced appropriately. Thermoflex insulation is placed on the inside of the metal skinning. In an emergency the whole of the canopy, complete with the central support member is jettisoned by firing explosive bolts.

6.4 Fin Structure

The maximum fin limit load of 56,300 lbf. occurs during sinusoidal rudder operation whilst the aircraft is flying at a Mach number of 0.95 at 60,000 ft. altitude.

The structure of the fin is mainly built up from components manufactured in D.T.D. 5070A, but some parts are made in S99 steel. It is based on a two cell box, the three spars being connected to fuselage frames at their root ends. Four major, chordwise, ribs carry hinge brackets for the rudder and three subsidiary ribs are placed between them. The resulting mean pitch is approximately 14 inches. The skin thickness varies between 0.05 inches at the tip and 0.11 inches at the root. Zed section stringers, 0.9 inches deep, are used to stiffen the skin.

A built up construction of plate webs and extruded tee section booms is employed for the front and centre spars. The rear spar is shallow and is a one piece machining in S99. The skins terminate at the root rib which is a complex machined component coincident with the top surface of the fuselage. The spars are kinked and have a manufacturing joint at this channel section rib. It also carries the lowest of the rudder hinges. The other hinge ribs are built up from pairs of 18g channel pressings placed back to back. Forged fittings are located on either side of the rear spar to transmit the hinge loads into the main box. The intermediate ribs consist of single channel pressings. Cutouts in the webs of all but the root and tip ribs are used to enable the stringers to be continuous, but these are joggled over skin-rib boom reinforcing straps.

The nose ribs are placed normal to the leading edge at half the pitch of the ribs in the main box. The 16g leading edge skins are further reinforced by 22g intercostal spanwise stiffeners.

6.4.1 Rudder

The maximum rudder limit load is 10,110 lbf. It arises during sinusoidal movement at a Mach number of 1.5

A full depth light alloy honeycomb construction is employed in the rudder. The skins are chemically etched in D.T.D. 5070A to a thickness which varies from 0.03 inches at the tip to 0.08 inches at the root. The leading edge is built around an extruded channel section spar to which the hinge fittings are attached. A triangular extrusion completes the trailing edge. There are two complete ribs and a partial one. The complete ribs are placed at the root and tip and are pressed channel sections. The partial rib assists the root rib in diffusing the control input torque loads.

6.5 Tailplane Structure

The maximum limit load on the complete tailplane in a symmetric manoeuvre is 80,000 lbf. which occurs in various symmetric manoeuvres. However in a combined pitching and rolling condition at sea level with a flight Mach number of 1.36 and a normal acceleration of 5.33g the limit load on one half of the tailplane can reach 55,000 lbf.

Since the tailplane is used for roll as well as longitudinal control each half is independently mounted on spigots which extend out from the sides of the fuselage. The main structural box in each part has a maximum width of 2.7 ft., with four equally spaced spars. The root 1.7 ft. is unswept and is used to transmit the bedding and shear loads into the two pivot bearings. Torsion loads are reacted on the control actuator attachment which is located on the root rib at the forward edge of the box. Outboard of this root region the box is sweptback at a mean angle of 26°. The box skin thickness varies from 14g at the root to 16g at the tip. These skins are supported by 1.25 inch deep 14g zed stringers located at a mean pitch of 2.7 inches between the spars. Both the skins and stringers are fabricated in D.T.D. 5070A.

The inner and outer bearings are carried on two substantial ribs which are machined from S99 forgings, the two central spars being discontinuous at the outer of them. The inner bearing is 7.0 inches in diameter and is tapered to react thrust loads. It has a glass fibre backed P.T.F.E. bearing surface mated with a chrome steel bush which is clamped to the spigot. The outer bearing unit consists of a pair of opposed tapered roller assemblies. The tailplane is locked on to the spigot with a ring nut at the outer bearing. Access to this is through a removable panel in the lower skin of the box.

The spars are built up from plate webs and angle section booms in D.T.D. 5070A. Subsidiary ribs are located at approximately 2 ft. pitch across the span and are channel section pressings, also in D.T.D. 5070A. The remainder of the tailplane structure consists of extensions to the ribs in the main box which are supported by a number of intercostal spar webs and supplemented by local 18g angle stiffeners. These, and the 16g covers, are in D.T.D. 5070A.

The aeroelastic characteristics of the tailplane have been investigated separately from the main project study by Ramsey (14) and Ibrahim (15).

These studies did not yield conclusive information due to the complexity of the problem relative to the digital computing facilities available.

6.6 Undercarriage

The tricycle undercarriage has a relatively narrow track which results from the necessity of mounting the undercarriage off the fuselage. Each of the legs carries two side by side wheels.

6.6.1 Main Undercarriage

The total stroke of the main undercarriage liquid spring shock absorbers is 25 inches. In a proof vertical descent of 20 ft/sec. at the maximum landing weight of 50,000 lb. the vertical reaction factor is 4.3 which corresponds with a vertical load of 107,500 lbf. on each leg. The general layout of the unit is shown in Figure 27.

The cantilever main leg has no support struts as such, the drag and side loads all being reacted at the top attachments. Whilst this results in a heavy unit it was found to be the only acceptable solution when retraction was considered. In the retracted position the legs and wheels are housed in two bays located beneath the powerplants. These bays are shared with the inner wing trailing edges when the wings are swept back. Space is at a premium, especially as it is necessary to cater for undercarriage extension or retraction with the wings in this position for maintenance on board a carrier. The retraction path is defined by the inclination of the main pintle skew hinge. The pintle is forged integrally with the top of the leg casing in FV520 steel. This component is welded to the lower part of the leg casing. The upper forging also incorporates a 10 inches long lever which registers with a lug on the main structure when the leg is extended and is locked by the downlock pin. Both the downlock and pintle brackets are located on the lower rear surface of the main wing box. The retraction jack is also attached to the top of the leg and is positioned between this and the fuselage keel member in the wheel bay. The connection of the jack to the leg is such that the lug is required to rotate about its own axis during retraction and hence it incorporates a fitting with a pair of tapered roller thrust bearings. A standard self-aligning unit is adequate for the keel end of the jack.

The axle fitting is bolted across the lower end of the sliding tube member. As well as providing attachments for the brake back plates it incorporates lugs for the lower torque links and the uplock pin. When the unit is retracted the uplock pin engages with a lug attached to a beam placed across the roof of the bay. The uplock is spring operated and hydraulically released. Twin plate disc brakes are fitted. The torque links are forged in L65 and attach to lugs on the bottom of the leg casing as well as those on the axle.

6.6.2 Nose Undercarriage

The design of the nose undercarriage unit was influenced considerably by the catapult take off condition. The maximum loads are 86,500 lbf. vertical and 21630 lbf. side in a three point landing and 66,000 lbf. drag in the high drag case. In the catapult condition the drag strut is subjected to 270,000 lbf. and the locked torque links to 85,000 lbf.

The unit is of cantilever design with a single drag strut which connects the axle fittings to the fuselage keel member. The twin wheels are carried on a live axle. Also attached to the axle fitting are the nosewheel tow catapult hook and the lower pivot for the unusually large torque links. The upper torque link pivot is attached to the fuselage structure rather than the leg casing. When the aircraft is in the catapult configuration the torque links are locked into a straight position and together with the radius rod drag strut form a rigid triangular structure which relieves the leg itself of all loads. The drag strut breaks for retraction and the unit is moved rearwards and upwards into the bay. The large, upper torque link acts as the forward bay door and because the pivot is offset from the main pintle hinge there is a small amount of preshortening.

The shock absorber is a liquid spring. The whole unit is built up from machined forgings in FV520 steel.

6.7 Fuel System

There are basically two complete fuel systems in the aircraft, one of which feeds the port and the other the starboard engine. There are six integral tanks on each side of the aircraft. One of these acts as a collector tank and is located in the fuselage just forward of the main wing box. Four of the other tanks are also in the fuselage and the remaining one is positioned in the leading edge of the inner wing. One complete system is shown in Figure 28.

The tanks are made fuel tight with a fluorosilicone sealant which is both caulked in the joints of the structure and then film sprayed over the entire inside surface. Access to the inside of the tanks is gained through removable panels which are located in the sides of the nosewheel bay, the web of the front spar and the rear walls of the mainwheel bays. The pair of rear tanks is located below the jet pipes. Tank insulation in this region consists of a 0.6 inches wide air gap formed by the corrugated titanium construction supplemented by Thermoflex blankets.

In each system the fuel is transferred to the collector tank by A. C. booster pumps. There are seven of these in each system as they are duplicated in the two tanks where gravity feed into the collector is not possible. The pumps are controlled by float switches placed in the collector box, by the displacement type fuel proportioners, or manually. Tank selection is used to adjust the centre of gravity to cater for the

variation required when the wing is swept.

In the most severe conditions of flight at Mach 1.4 at sea level each engine requires 5,600 gallons per hour. This is supplied from the collector tank through a fuelhydraulic feed system. Either of the two fuelhydraulic booster pumps in each system is capable of meeting the requirements. The design provides for a cross feed between the two systems between the collector tanks and the power plants, and transfer between the collector tanks themselves to cater for various failure conditions. These include damage to a collector tank as a result of enemy action, in which case it would be isolated completely. However the collector tanks are positioned in a region which contains substantial structural members and additional armour plating could be arranged to make damage of this type extremely unlikely.

The aircraft would normally be refuelled through a single 2.5 inch valve positioned in the nosewheel bay. At a rate of 500 gallons per minute the aircraft can be refuelled from empty to full in 5.7 minutes, the capacity of the system being 22860 lb. Provision is made for nitrogen injection during pressure refuelling and gravity filling to individual tanks. There are separate galleries for each system and these are reduced in bore towards their ends to control the rate of filling to each tank. There are also float switches and pressure relief valves.

10000 lb of fuel can be jettisoned in 7.67 minutes from a single outlet placed under the rear fuselage. The jettison pipe is connected to the main feed lines through control valves. The venting and pressurising system are combined. A nitrogen-air mixture is used to maintain the tank differential pressure between 2 and 2.5 p.s.i. The vent outlet is located in the same region as the jettison pipe. Capacitance type fuel contents gauging is used, there being 20 tank units in each half of the system. Although fuel flowmeters of the capacity required are not available, provision has been made for them in the main feed lines.

6.8 Powerplant Installation

The two 15,000 lbf. sea level static thrust bypass engines are positioned in the rear fuselage, immediately aft of the rear spar. They are installed through large holes in the upper surface of the fuselage.

Two dimensional, wedge type air intakes are located on the upper surface of the fuselage, some distance aft of the cockpit. The air intake ducts change from a rectangular to circular cross section as they pass through the centre fuselage. Spot welded light alloy construction is used for these and they are supported by the fuselage frames. It is envisaged that titanium alloys would be used for the intake wedges.

The jet pipes terminate in rectangular section, two dimensional, variable area convergent divergent nozzles. These are supported off the end of the fuselage structure. Insulation is provided between the jet pipes and the roof of the integral fuel tanks below them.

7.0 Discussion

7.1 Hinge Systems

The weights of the two hinge systems considered for the design were estimated using similar assumptions. There was no definite indication that, as designed, one arrangement was any lighter than the other. However it should be noted that the structural and geometric layout of the aircraft favoured the use of a horizontal shear type of hinge. There is little doubt that it would be possible to revise the layout so that it would be more suited to the use of a hybrid hinge system, and in this case a lighter design could be expected. Against this possible lower weight of the hybrid system must be placed the advantages of the alternative scheme of smaller volumetric requirements and more straightforward fairing and general installation. In general, therefore, there is probably very little to choose between the two types, and certainly in the present design the horizontal system is preferable.

7.2 Roll Control

An unusual feature of the design is the method of obtaining roll control. Whilst independently moving tailplanes have been used for this purpose on one or two other designs, the difference in the case of the S64 is the need for additional low speed roll control to meet the naval deck landing requirements. Spoilers are not a good solution to this difficulty since the real need is to be able to lift one wing without an overall loss of lift, and hence differential flap movement was suggested. Whilst the flaps are capable of providing the necessary lift increments the major problems are ones of response and adverse yaw effect. The former requires the replacement of the normal flap actuators by fully powered control units which have to be integrated with the normal roll control system. In retrospect it would seem that some other alternative, possibly some form of superimposed circulation control, might be a more realistic solution to this problem.

7.3 Nosewheel Tow

There has been very little experience of designing aircraft for nosewheel tow catapult launch and even less actual operating experience. The loads imposed upon the nosewheel are inevitably large, but this is a penalty which must be expected, and accepted, to gain the improved deck operation of the aircraft. Of more specific significance is the proposal that for the nosewheel tow condition the S64 nose undercarriage should effectively be locked as a rigid triangulated structure. This prevents the shock absorber from contracting and relieves the loads on it during the launch phase, at the expense of very heavy torque links.

Whether this is the correct approach, or even whether it is feasible, could probably only be determined by full scale trials.

7.4 Cockpit Layout

The unusual inclination in plan of the ejector seats has been used to obtain side by side seating within a minimum fuselage width. An alternative way to achieve the same end would be the elimination of the individual ejector seats in favour of an escape capsule, but this was considered to require considerable development. A full scale mock up and probable flight trials would be necessary to decide if the suggested inclined seats are acceptable, but inclined steering wheels are commonly used in road vehicles.

7.5 Engine Installation

The low wing of the aircraft together with the fully buried engines results in a difficult engine access problem. The possible solutions to this are the removal of the engines aft through the jet pipe tunnels, the disconnection of the complete rear fuselage, or the provision of large upper fuselage access panels. The latter method was used as the structural penalties were found to be acceptable. Adequate bending material is available in the fuselage sides, keel member, and the vertical and horizontal shear webs. The torsion strength problem was overcome by using shear carrying quick release fasteners of a type used successfully on guided missiles. The major objection is the height required to lift the engine clear of the fuselage.

8.0 Conclusions

- (1) Although there is little to choose between the horizontal and hybrid shear systems considered for the design on a basis of weight, the former is preferable in that it gives more scope in the overall layout of the aircraft.
- (2) The lowspeed roll requirements cannot be met with independent tailplane movement, and the use of differential flap movement is probably not acceptable because of the poor control response and adverse yaw effect.
- (3) Nosewheel tow catapult launch incurs a considerable weight penalty. It is not clear whether the proposed method of locking the nose under-carriage for this is the correct solution.
- (4) The buried engine installation with large, quickly detachable, access panels is a satisfactory solution to the power plant installation problem.

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TABLE 1
WEIGHT BREAKDOWN
(Predicted)

Component	Weight lb.	% A. U. W.
Wing - Inner	3470	5.8
- Outer	6600	11.0
Fuselage	5540	9.2
Tailplane	780	1.3
Fin and Rudder	380	0.6
Main U/C	2200	3.7
Nose U/C	670	1.1
Structure	19640	32.7
Engines	4500	7.5
Installation	220	0.4
Intakes	480	0.8
Jet pipes and nozzles	1200	2.0
Power Plant	6400	10.7
Fuel system	1100	1.8
Flying controls	400	0.7
Hydraulics	850	1.4
Electrics	1500	2.5
Cabin systems	350	0.6
Radio and radar	1500	2.5
Instruments	500	0.8
Seats, etc.	400	0.7
Weapon carriers	140	0.2
Systems and Equipment	6740	11.2
Crew	360	0.6
Total zero fuel and payload	33140	55.2
Max. payload	4000	6.7
Total zero fuel	37140	61.9
Max. fuel	22860	38.1
Max. A. U. W.	60000	100.0

TABLE 2

Centres of Gravity and Moments of Inertia

	Case	Weight lb.	\bar{x} ft positive aft of hinge datum	\bar{z} ft positive below hinge datum	Moments of Inertia about hinge datum lb ft ²		
					Roll	Pitch	Yaw
Undercarriage Retracted	Zero fuel and payload: high speed	33140	-0.52	-0.92	2.77×10^6	6.05×10^6	8.57×10^6
	Zero fuel and payload: high speed	33140	+0.20	-0.96	2.16×10^6	6.68×10^6	8.59×10^6
	Zero payload, full fuel: low speed	56000	Fuel disposed and used to keep c.g. as shown in Figure 7.		2.90×10^6	10.1×10^6	12.68×10^6
	Zero payload, full fuel: high speed	56000			2.28×10^6	10.73×10^6	12.70×10^6
	A. U. W. low speed *	60000			2.99×10^6	10.18×10^6	12.75×10^6
	A. U. W. high speed *	60000			2.38×10^6	10.81×10^6	12.77×10^6
Extended	Zero fuel and payload: low speed	33140	-0.63	-0.63	2.83×10^6	5.93×10^6	8.45×10^6
	A. U. W. low speed *	60000			3.05×10^6	10.06×10^6	12.63×10^6

* Payload assumed is 4 x 1000 lb bombs.

APPENDIX A

Allocation of Components for S64 Study

Barrett, W. F.	Fuel System
Bishop, W.	Wing hinge
Bright, D. M. F.	Rear fuselage
Clark, A. P.	Centre fuselage and wing
Coles, P. J.	Fuselage nose
Evans, G. J.	Undercarriage
McKenzie, W.	Front fuselage
Ramsey, R.	Tailplane
Read, J. M.	Fin and rudder
Withers, D. L.	Outer wing

Additional study in 1966-67 academic year

Noffsinger, J. L.	Alternative pin and track hinge.
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APPENDIX B

Specification of Aircraft

1.0 Geometry

1.1 Wing

The planform shape and hinge geometry is shown in Figure 13.

Aerodynamic nominal reference area 700 sq. ft.
 Low speed configuration:-

Gross area, actual	723 sq. ft.
Span	60.0 ft.
Aspect Ratio	4.98
Leading edge sweepback, inner panel	66°
outer panel	13°
Trailing edge sweepback	9°
Standard mean chord	12.05 ft.

Intermediate aerodynamic reference configuration:

Gross area, actual	695 sq. ft.
Span	51.4 ft.
Leading edge sweepback on outer panel	40°

High speed configuration:-

Gross area, actual	693 sq. ft.
Span	37.6 ft.
Aspect Ratio	2.05
Leading edge sweepback, whole wing	66°
Trailing edge sweepback	62°
Standard mean chord	18.4 ft.

Aircraft carrier folded configuration:-

Span	30.0 ft.
Wing angle to body datum	3°
Location of wing hinge below fuselage datum at hingeline	0.5 ft.
Location of hinge line aft of nominal centreline	
leading edge position	25.3
Location of hinge outboard of centreline	6.38 ft.
Location of hinge line aft of fuselage nose datum	38.8 ft.
Location of wing fairing fence outboard of centreline	9.57 ft.

1.2 Trailing edge flaps

Type: Double slotted (used in low speed configuration only)

Overall flap chord/wing chord	0.3
Inboard end of flap from aircraft centreline	9.6 ft.

Outboard end of flap from aircraft centreline	29.0 ft.
Take off flap angle	35°
Landing flap angle, maximum	65°
Differential flap angle for landing roll control	60° ± 5°

1.3 Ailerons

These are not fitted in the design, roll control being obtained by differential tailplane movement, supplemented at low speed by differential flap movement.

1.4 Tailplane

Gross area	220.7 sq. ft.
Span	25.0 ft.
Aspect ratio	3.0
Leading edge sweepback	45°
Trailing edge sweepback	0°
Nominal chord on aircraft centreline	15.13 ft.
Nominal tip chord	2.55 ft.
Standard mean chord	8.65 ft.
Location of tailplane datum above fuselage datum (tailplane angle neutral)	0.4 ft.
Location of pivot aft of fuselage nose datum	62.0 ft.
Location of pivot aft of nominal centreline leading edge position	10.3 ft.
Aerofoil section: Biconvex, 6% thickness chord ratio	
Movement	± 6°

1.5 Elevators

Elevators are not fitted.

1.6 Fin

Reference area	120 sq. ft.
Dorsal fin:	
Area above fuselage datum	120.8 sq. ft.
Height above fuselage datum	12.5 ft.
Aspect ratio	2.59
Leading edge sweepback	47°
Trailing edge sweepback	0°
Root chord at fuselage datum	16.32 ft.
Nominal tip chord	3.0 ft.
Standard mean chord	9.65 ft.
Location of leading edge at fuselage datum aft of fuselage nose datum	50.43 ft.
Aerofoil section: Biconvex, 5% thickness chord ratio.	
Ventral fin:	
Area below fuselage datum	9.0 sq. ft.

Depth below fuselage datum 2.75 ft.

1.7 Rudder

Type: Plain control, sealed hinge.

Area	25 sq. ft.
Location of root chord above fuselage datum	5.25 ft.
Location of tip chord above fuselage datum	12.5 ft.
Root chord	2.65 ft.
Tip chord	1.35 ft.
Hinge line sweepback	10°
Movement	±20°

1.8 Fuselage

Overall length	74.8 ft.
Length with nose and tail folded	57.2 ft.
Maximum width	9.5 ft.
Maximum depth	6.75 ft.
Overall aircraft height at 60,000 lb. weight	17.75 ft.

1.9 Undercarriage

Type: Nosewheel

Wheelbase (static configuration)	24.2 ft.
Track	10.0 ft.
Design vertical velocity (proof)	20.0 ft/sec.
Angle of take off ground line to fuselage datum	4°
Mainwheel tyres 30.0 ins. diameter x 7.0 ins. width	
Mainwheel tyres nominal pressure	250 p.s.i.
Nosewheel tyres 23.0 ins. diameter x 7.0 ins. width	
Nosewheel tyres nominal pressure	160 p.s.i.
Location of mainwheel axle aft of fuselage nose datum (parallel to fuselage datum)	40.8 ft.

2.0 Power Plants

Type: Twin bypass jet engines with provision for burning of additional fuel in the exhaust.

Sea level static thrust, cold, per unit	14,500 lbf.
Overall length	8.8 ft.
Maximum diameter	3.25 ft.

Air intake: Variable area, two dimensional three shock type based on adjusting wedge angle.

Location of intake edge at centreline aft of fuselage nose datum	21.75 ft.
Location of engine front face aft of fuselage nose datum	40.8 ft.

Exhausts: Two dimensional variable area convergent-divergent nozzle.

Exhaust pipe diameter 2.7 ft.

3.0 Weights

Maximum all up weight	60,000 lb.
Maximum landing weight	50,000 lb.
Maximum internal fuel load	22,860 lb.
Normal maximum payload	4,000 lb.
Zero fuel and payload weight	33,140 lb.
Weight breakdown, see Table 1.	
Centres of gravity, see Table 2.	
Moments of Inertia, see Table 2.	

4.0 Aerodynamic Characteristics

Wing-body lift curve slope, a_1	
Low speed configuration, incompressible	4.4/rad.
High speed configuration, $M = 1.0$	2.1/rad.
$M = 2.0$	1.9/rad.
Tailplane lift curve slope, a_{1T} , incompressible	2.5/rad.
$M = 1.0$	2.8/rad.
$M = 2.0$	1.8/rad.
Fin lift curve slope, a_{1F} , incompressible	2.8/rad.
$M = 1.0$	4.0/rad.
$M = 2.0$	2.3/rad.
Ratio of rudder to fin lift curve slopes, $\frac{a_{2F}}{a_{1F}}$, low speed	0.52/rad.
Variation of wing-body aerodynamic centre with Mach No. see Fig. 7	
Tailplane aerodynamic centre, low speed, aft of hinge datum	22.5 ft.
$M = 2.0$, aft of hinge datum	24.3 ft.
Fin aerodynamic centre, low speed, aft of hinge datum	20.1 ft.
$M = 2.0$, aft of hinge datum	22.4 ft.
Pitching moment coefficient at zero lift, low speed	-0.016
Rudder hinge moment coefficient due to fin incidence, b_{1F} , low speed	-0.25
Rudder hinge moment coefficient due to rudder angle, b_{2F} , low speed	-0.40
Maximum lift coefficients, untrimmed	
Basic wing	0.94
Flaps at 35° , take off position	1.55
Flaps at 60° , landing position,	1.75
Take off lift coefficient	1.32

Rolling moment coefficient due to sideslip, l_v , lowspeed	$-(0.04+0.07C_L)$
Rolling moment coefficient due to roll, l_p , lowspeed	-0.49
Rolling moment coefficient due to yaw, l_r , lowspeed	$0.027+2.44C_L$
Yawing moment coefficient due to sideslip, n_v , lowspeed	0.12
Yawing moment coefficient due to yaw, n_r , lowspeed	$-(0.109+0.018C_L^2)$
Rolling moment coefficient due to differential tailplane movement, l_ξ	-0.076
Rolling moment effect due to differential flap operation at landing, $l_{\xi F}$	-0.140

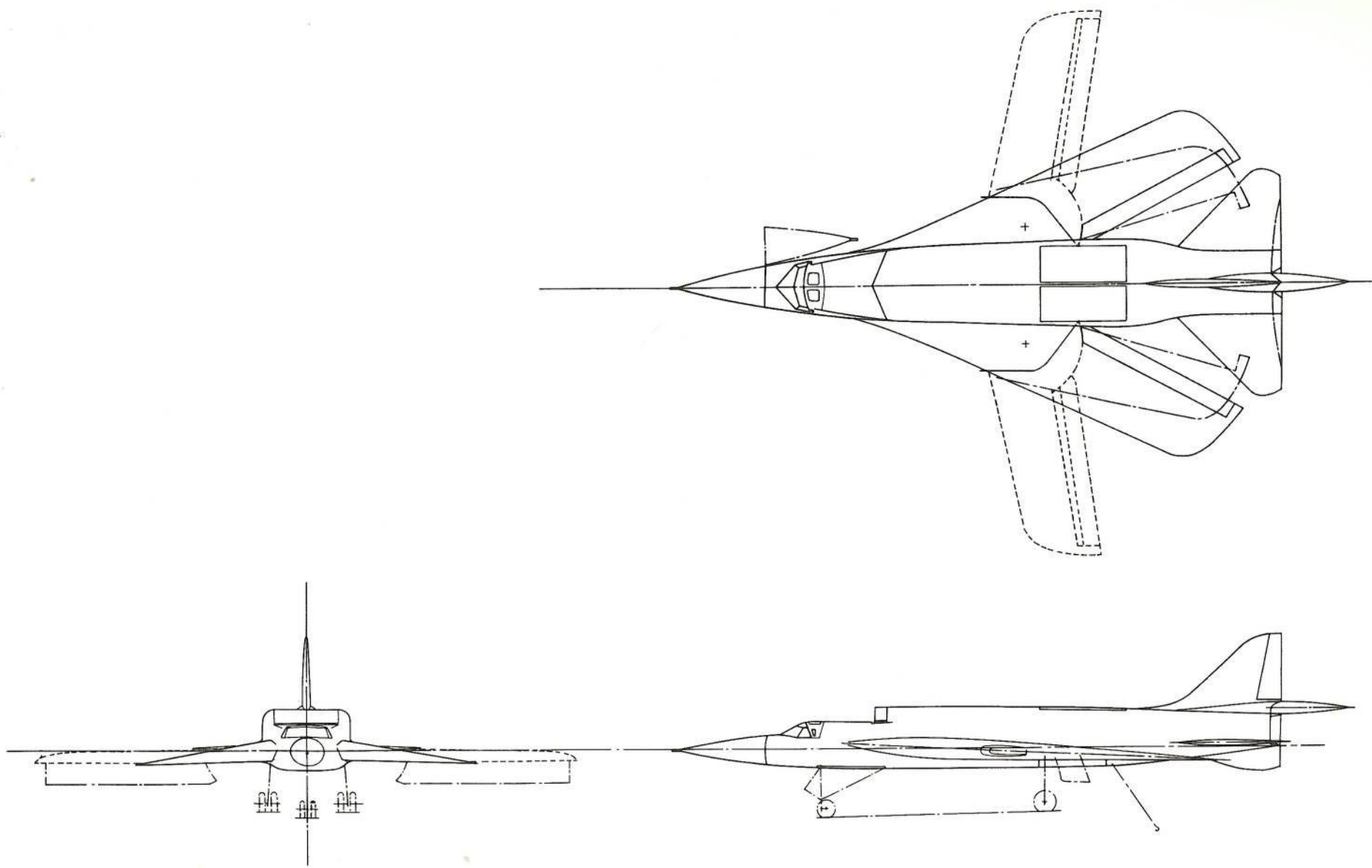


FIG.1. G.A. OF S 64 NAVAL STRIKE AIRCRAFT

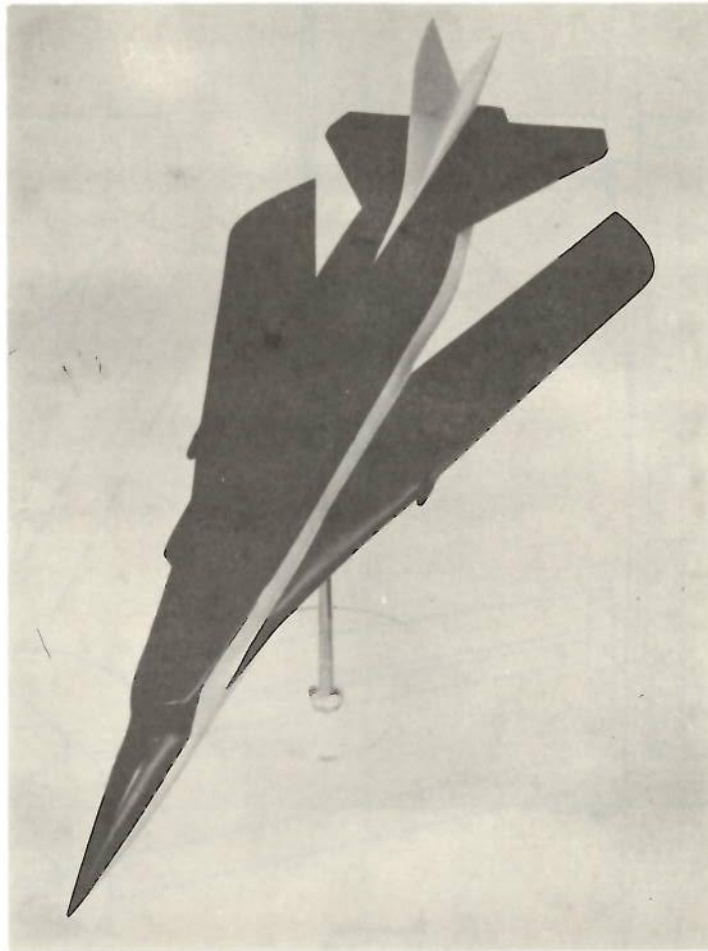


FIG. 2. PHOTOGRAPH OF A MODEL

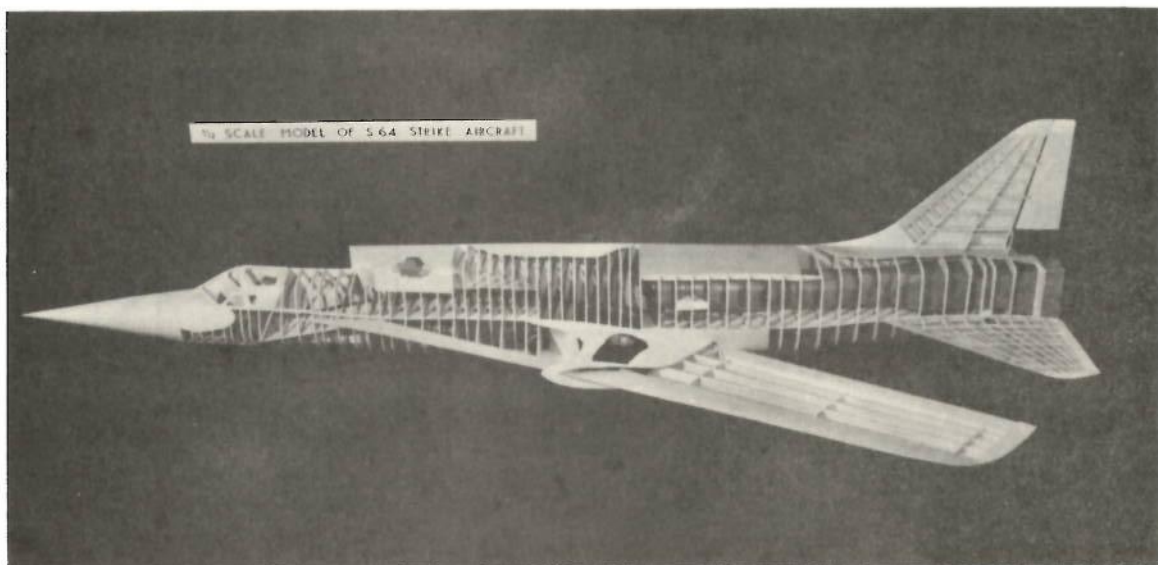


FIG. 3. PHOTOGRAPH OF A STRUCTURAL MODEL

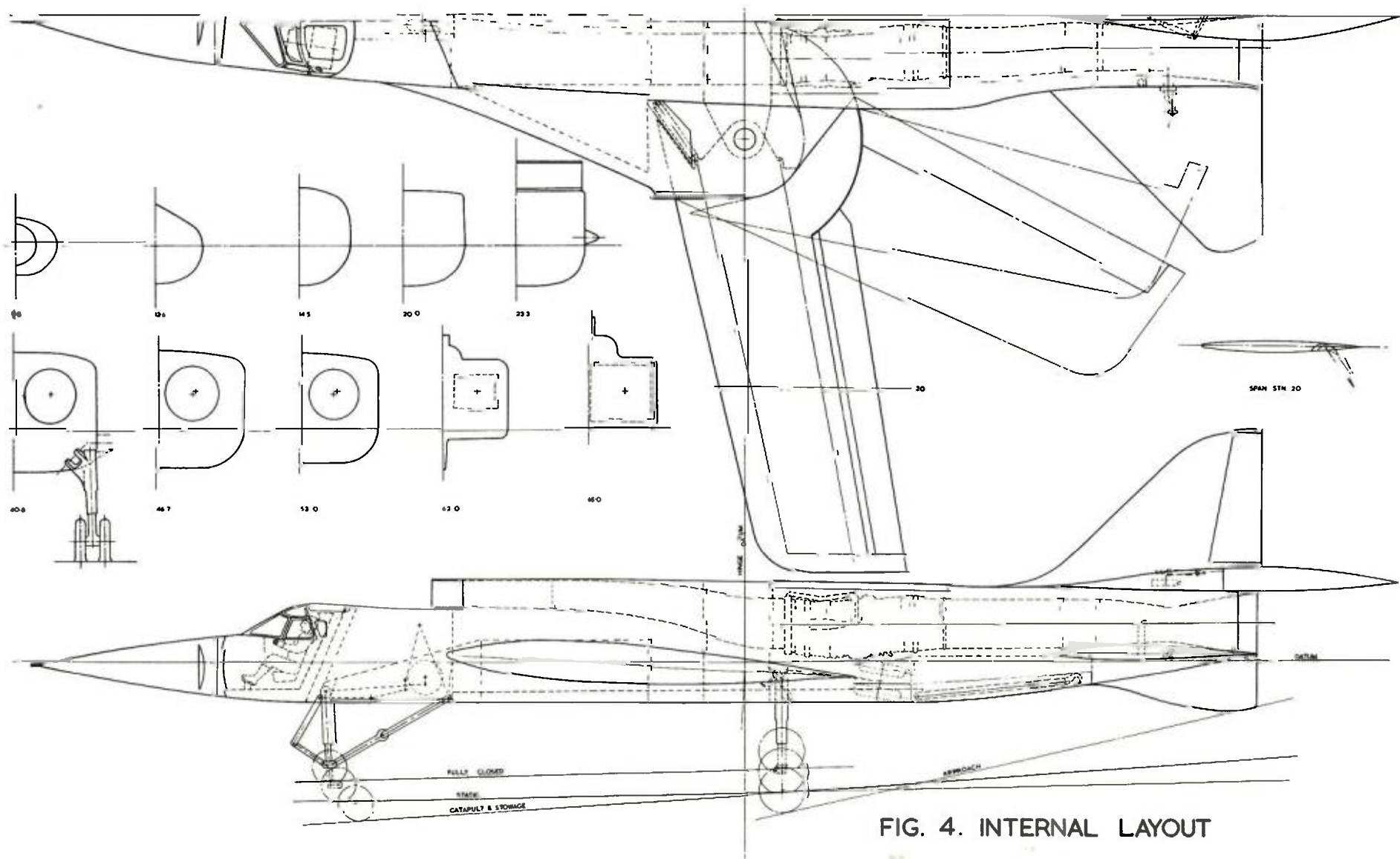


FIG. 4. INTERNAL LAYOUT

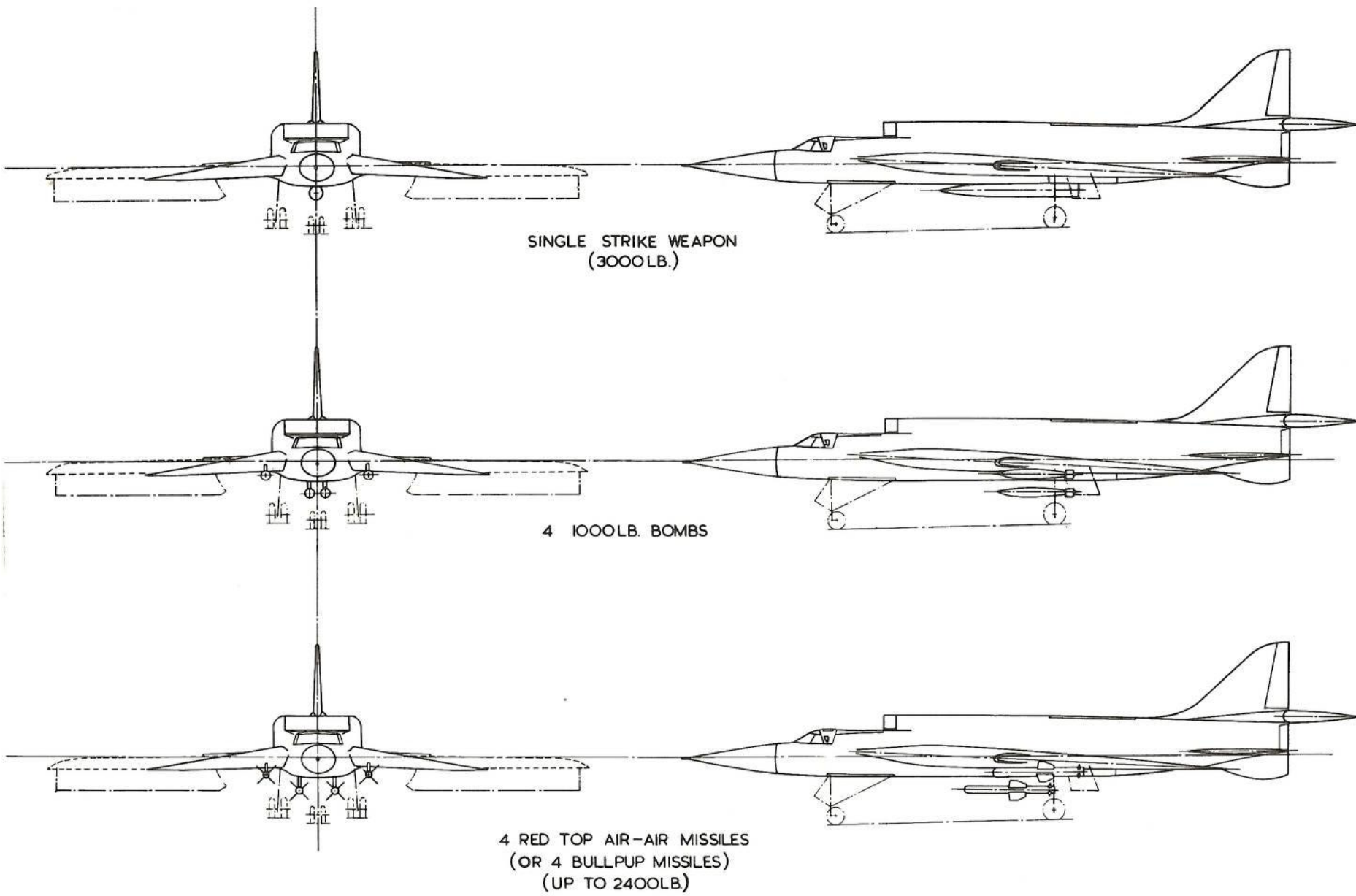


FIG. 5. TYPICAL WEAPON LOADS

AIRFRAME FLIGHT LIMITATIONS

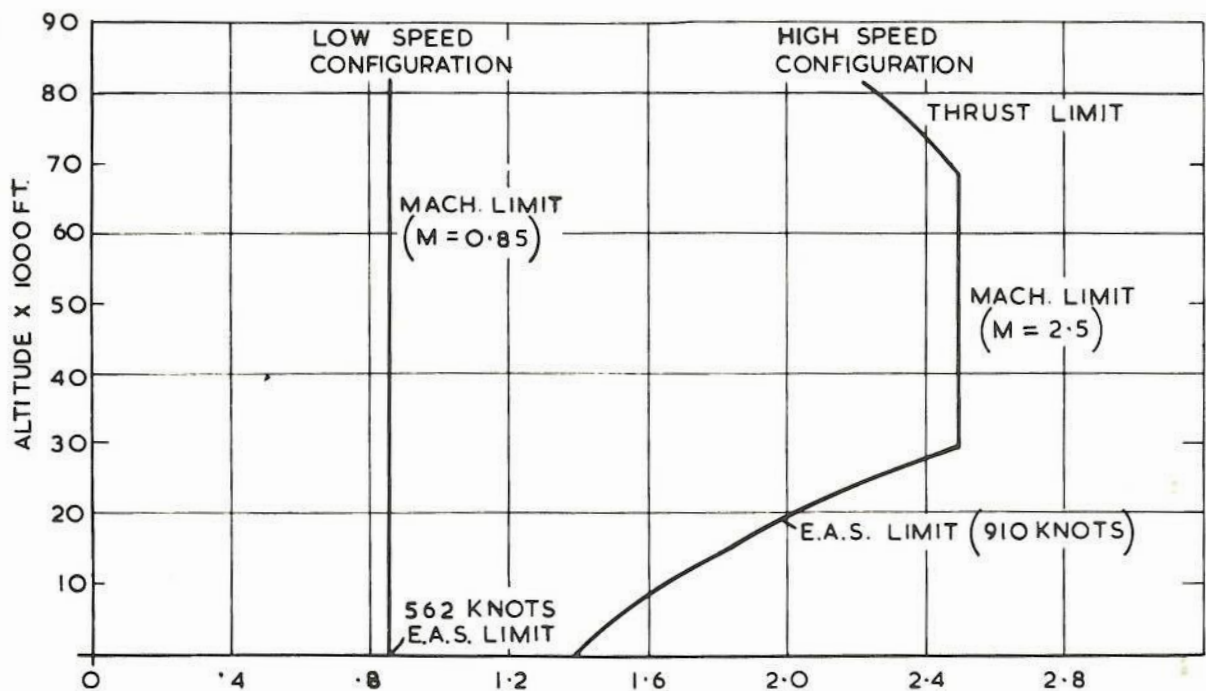


FIG. S64 - 31

FIG. 6. DESIGN FLIGHT LIMITATIONS

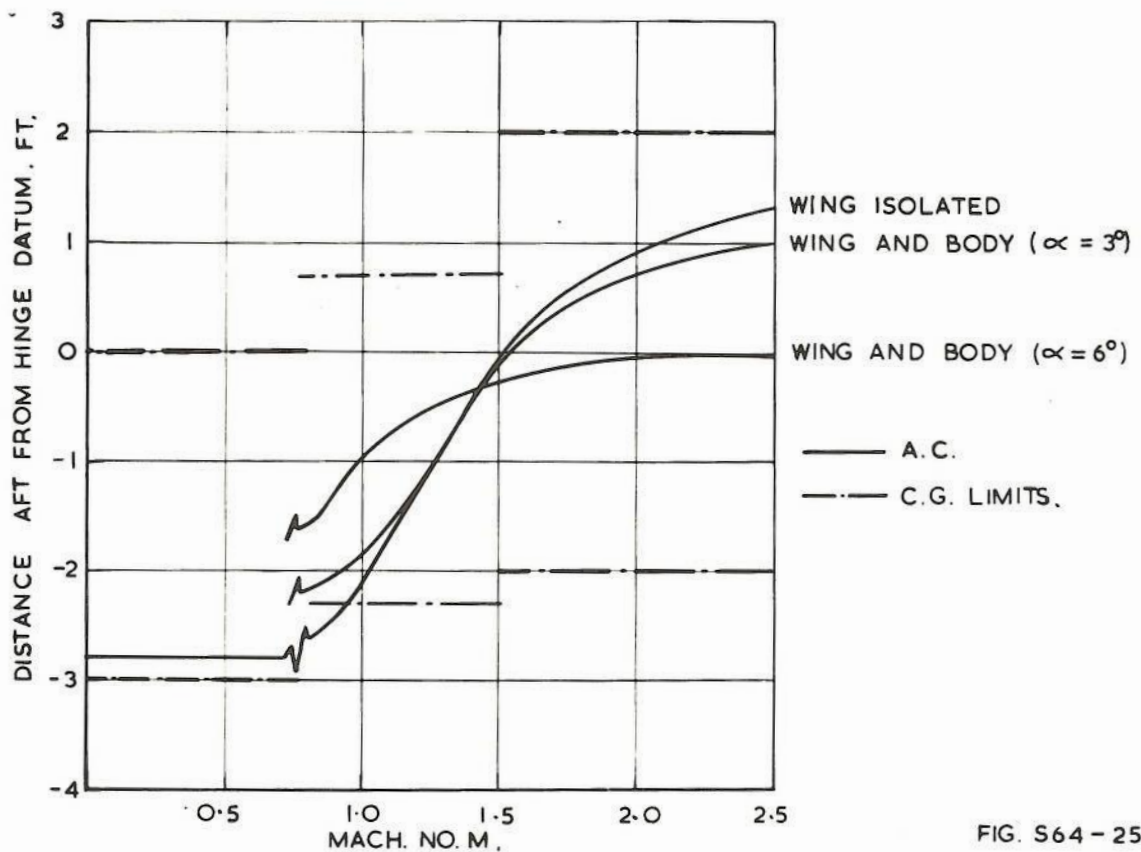


FIG. S64 - 25.

FIG. 7. AERODYNAMIC CENTRE AND CENTRE OF GRAVITY VARIATION

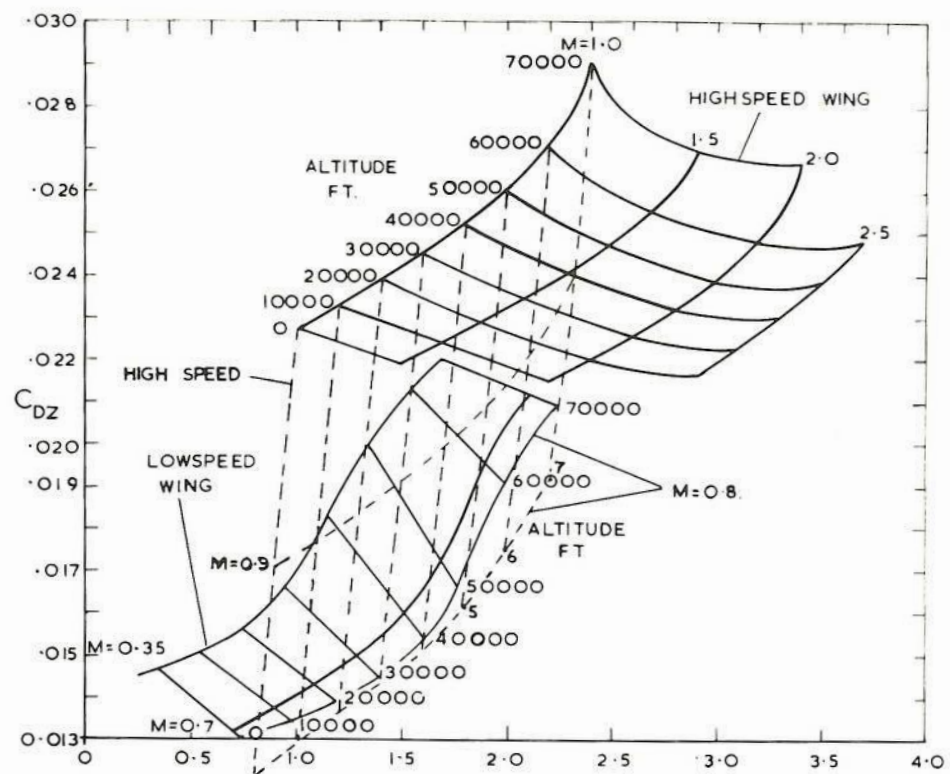


FIG. 8. ESTIMATED PROFILE DRAG COEFFICIENTS

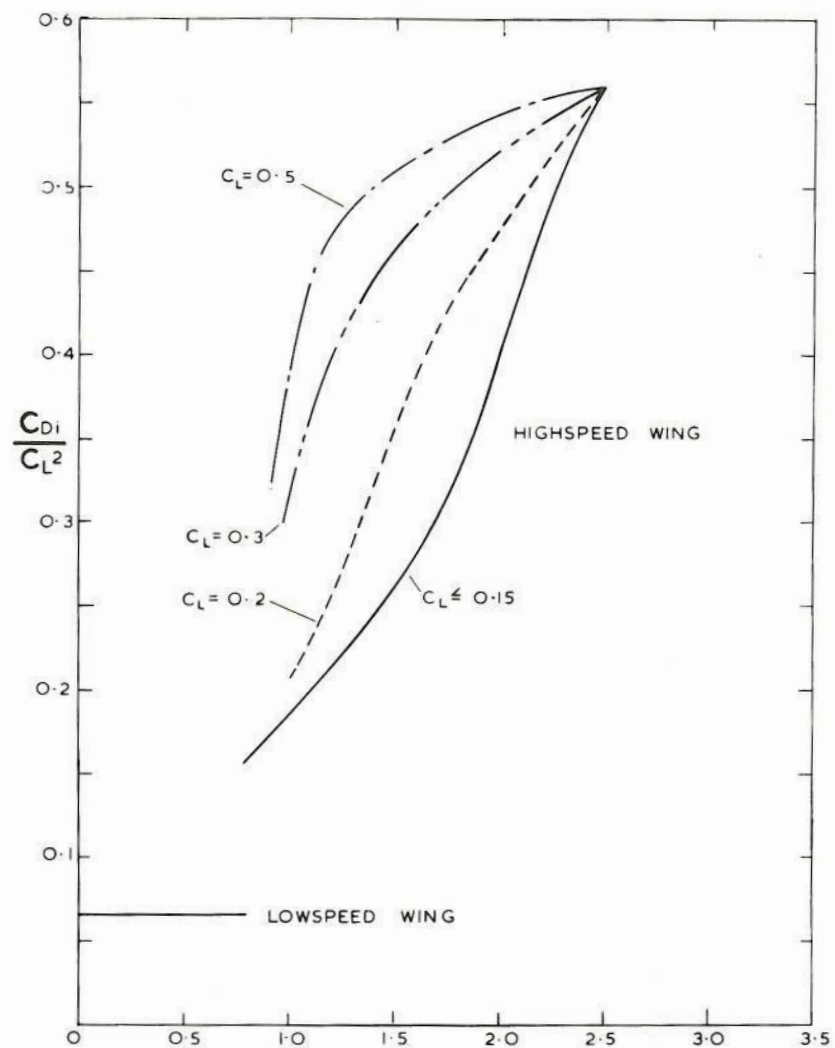


FIG. 9. ESTIMATED INDUCED DRAG FACTORS

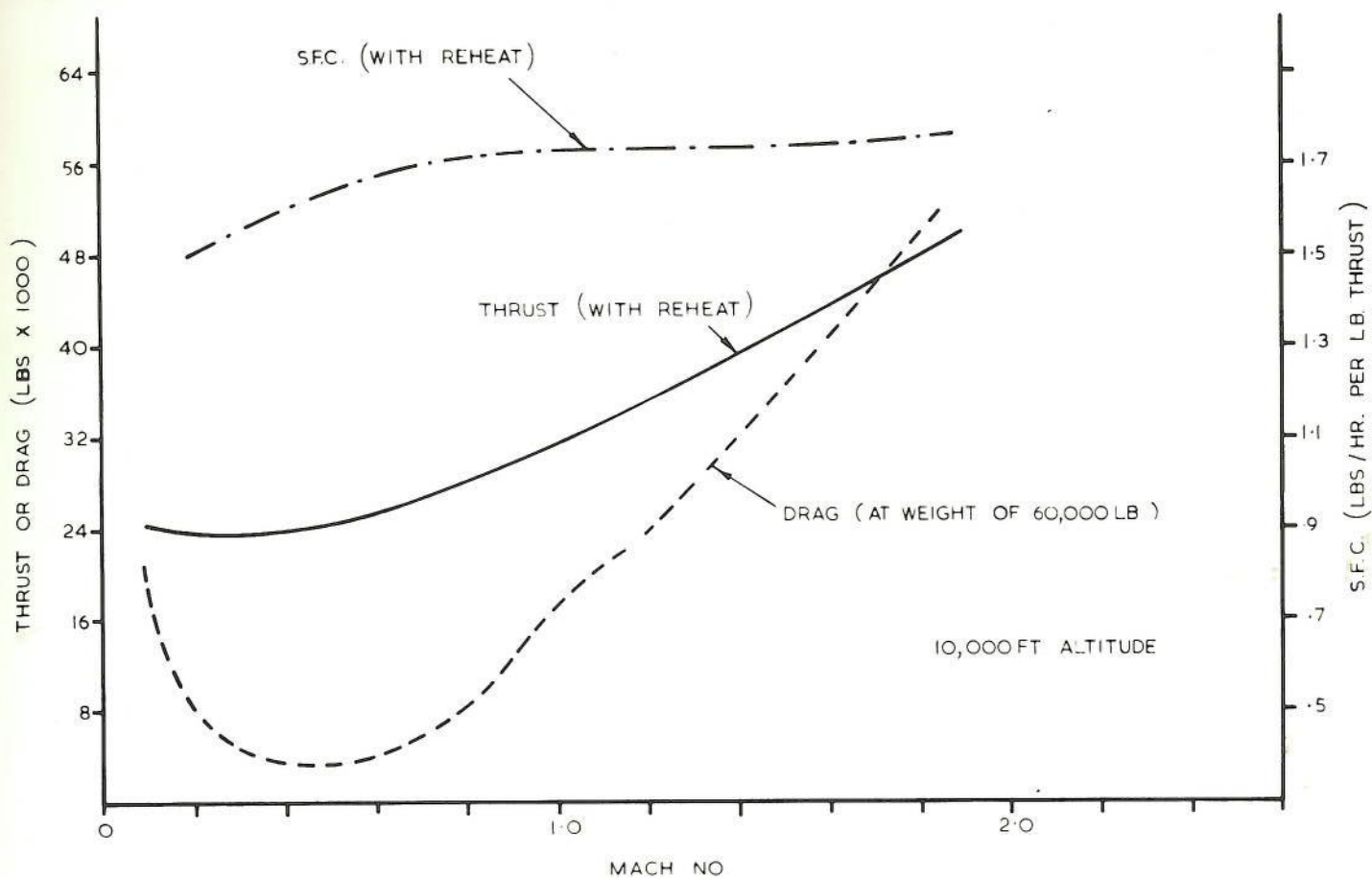


FIG. 10. THRUST AND DRAG VARIATION

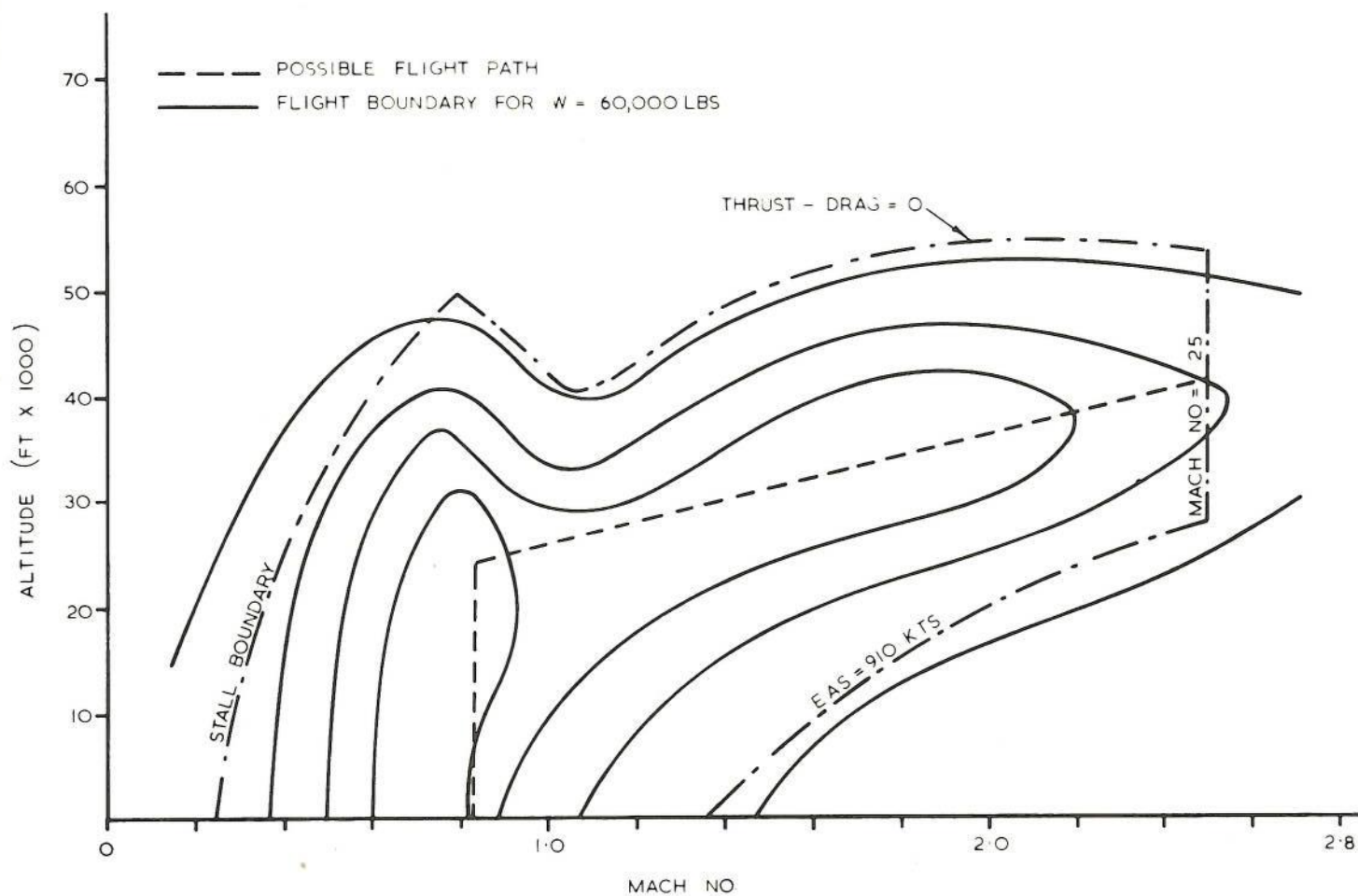


FIG. 11. ENERGY HEIGHT CHARACTERISTICS

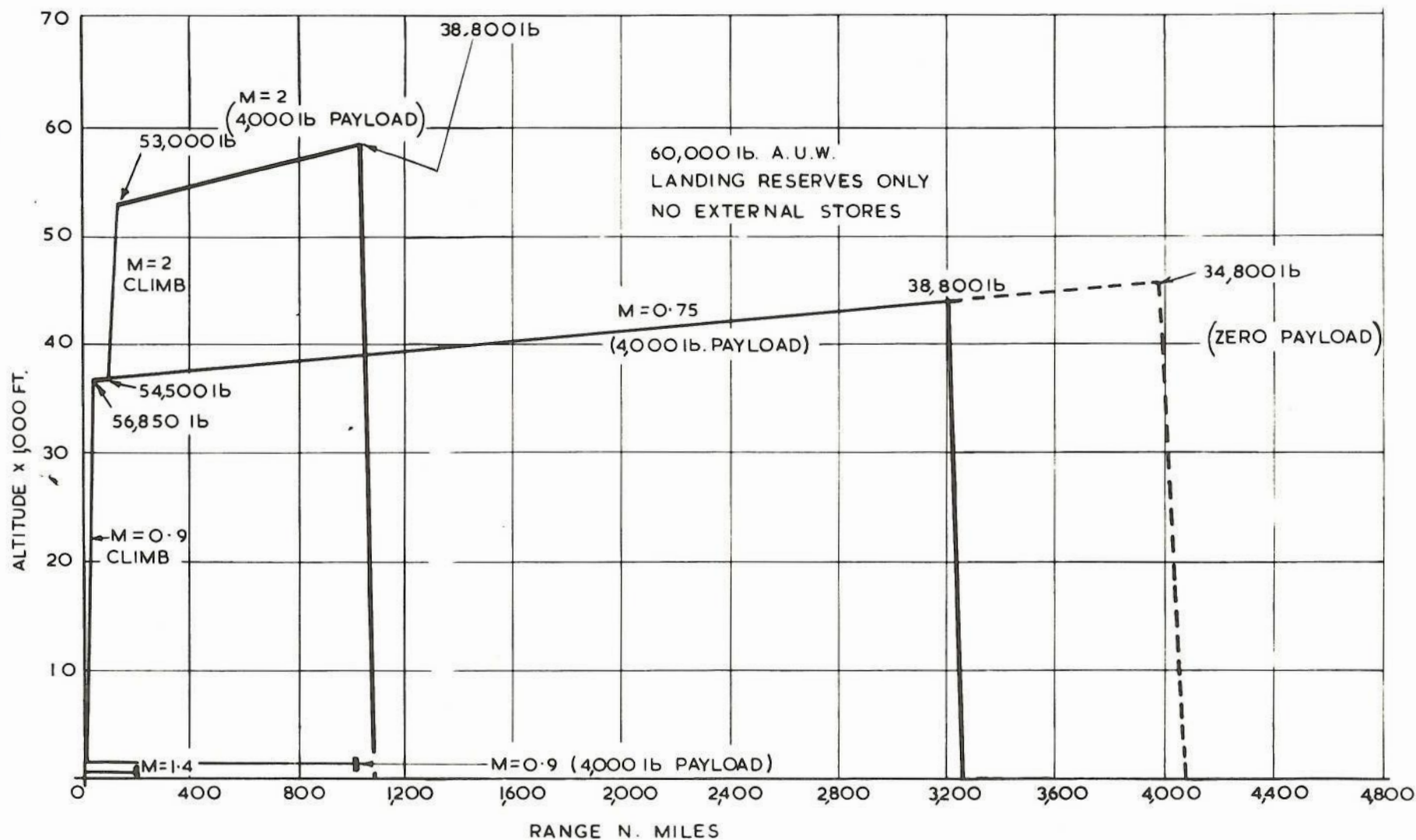


FIG. 12. TYPICAL RANGE PERFORMANCE

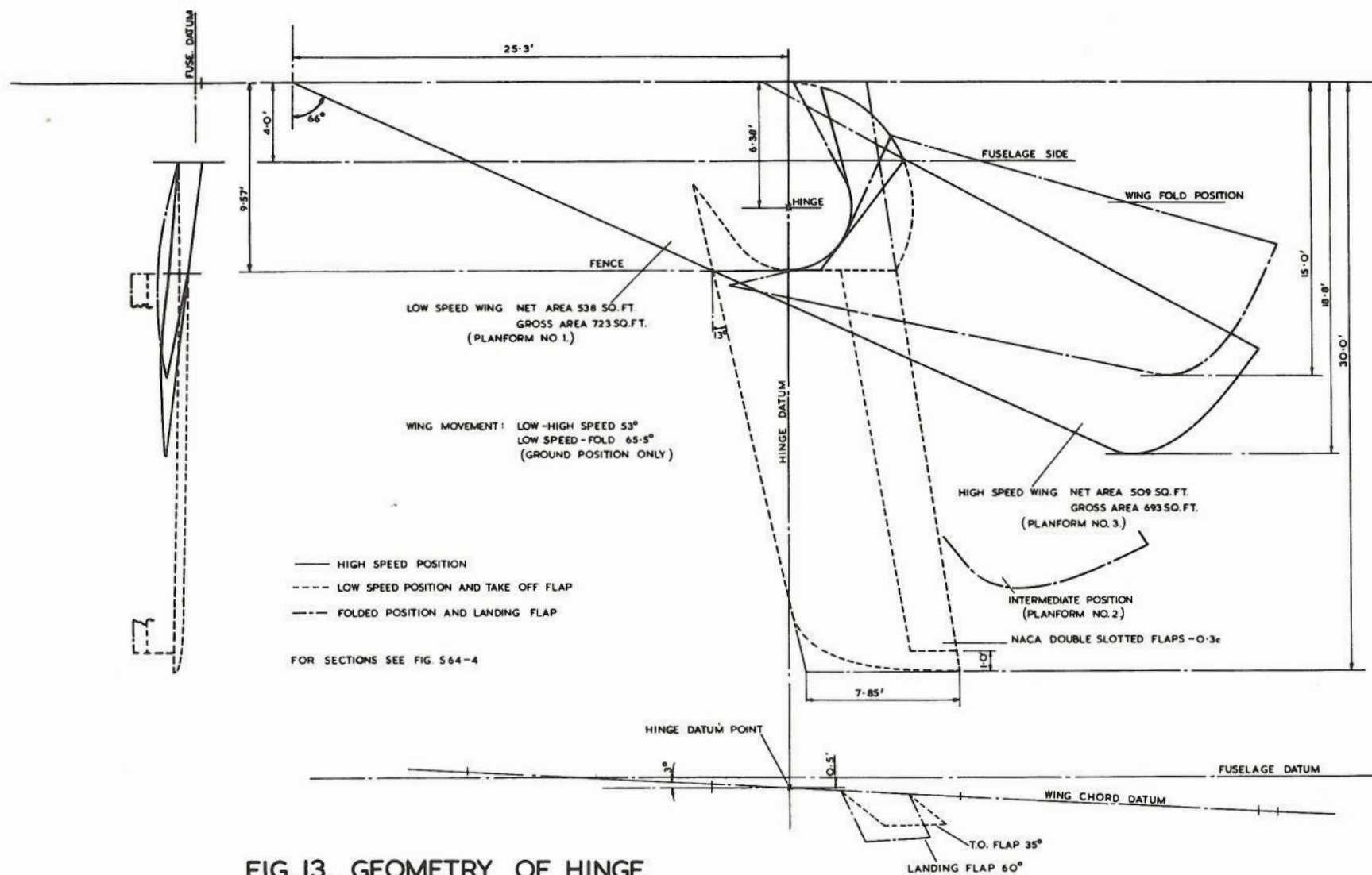


FIG. 13. GEOMETRY OF HINGE

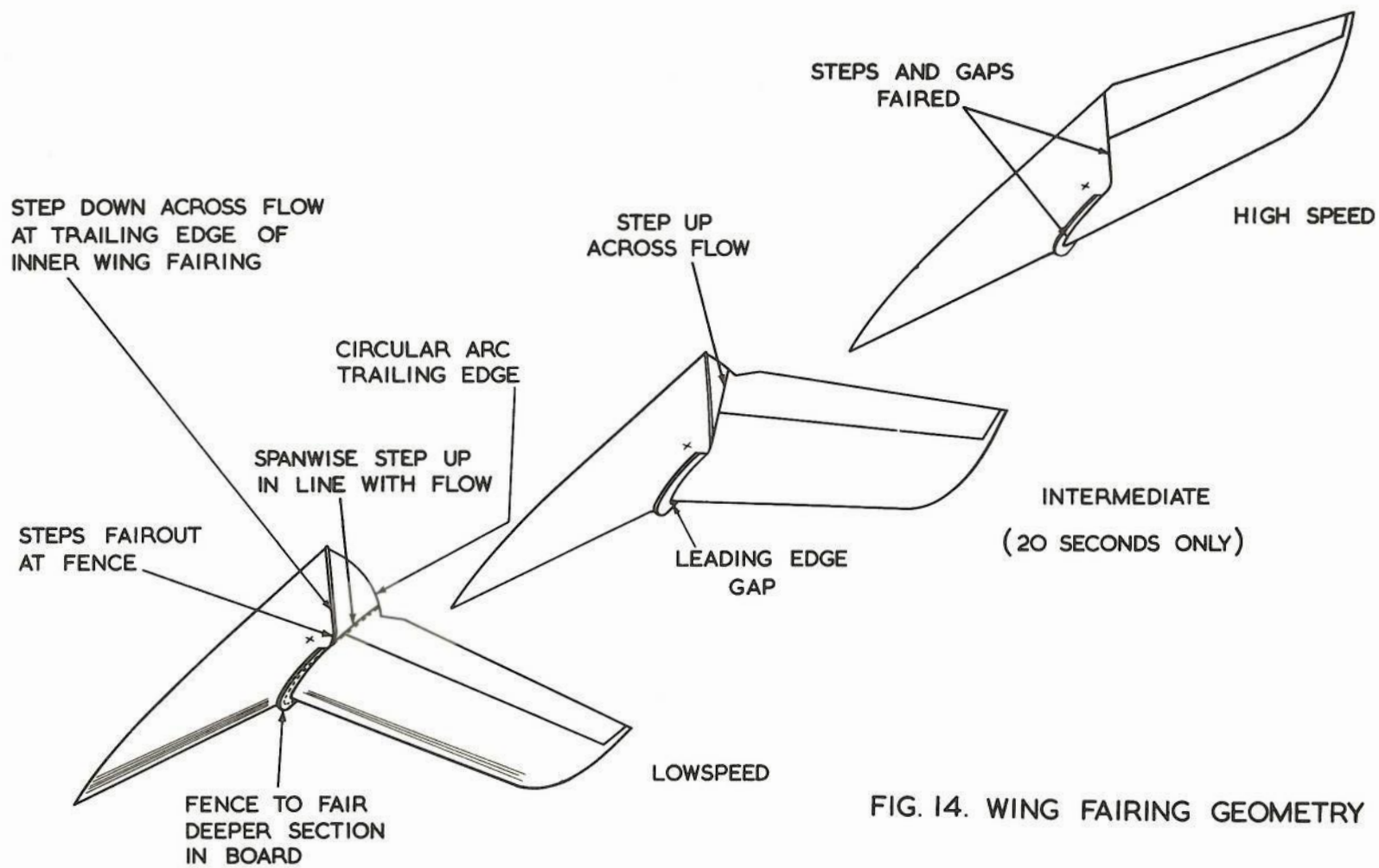


FIG. 14. WING FAIRING GEOMETRY

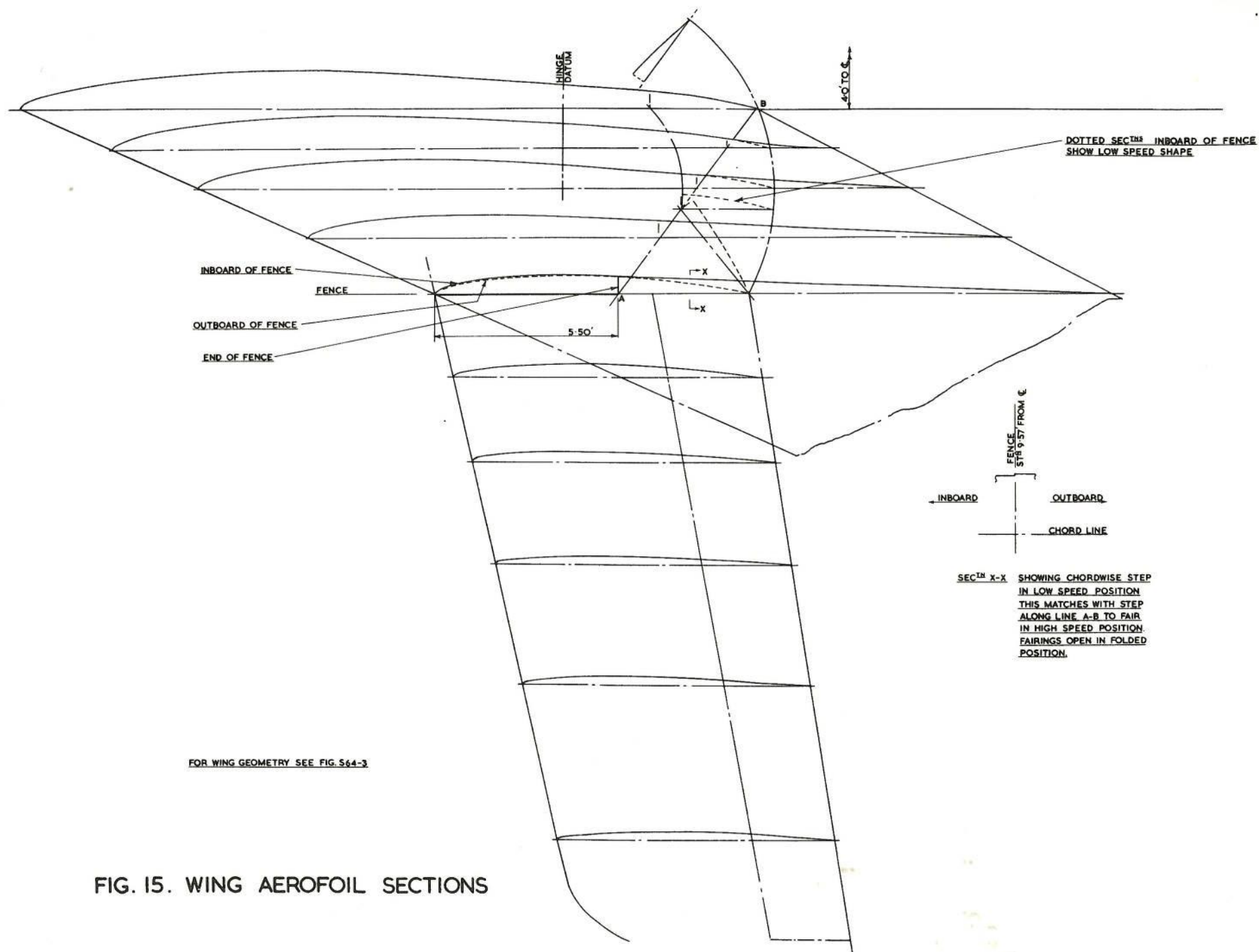


FIG. 15. WING AEROFOIL SECTIONS

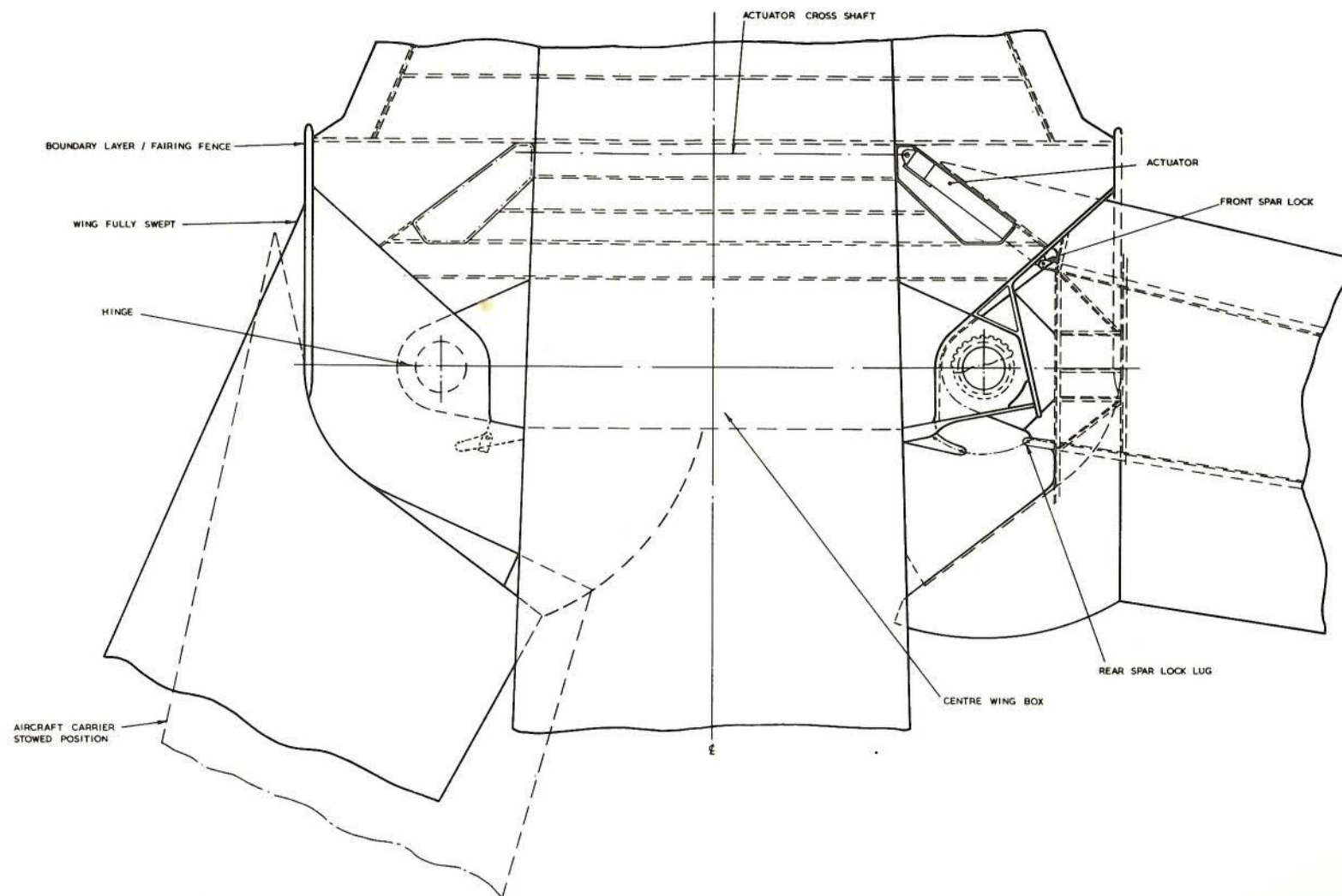


FIG. 16. G. A. OF HORIZONTAL SHEAR HINGE SYSTEM

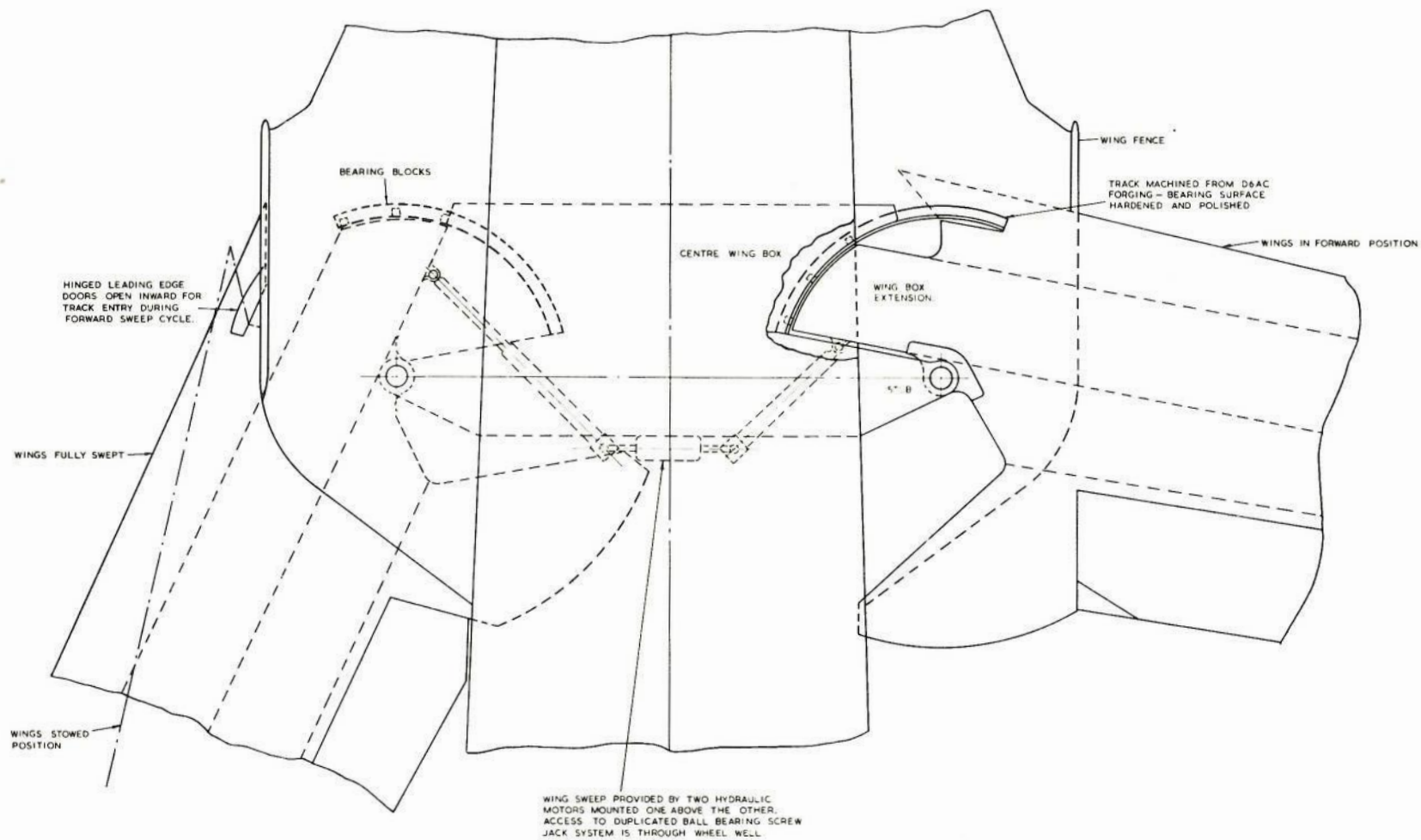


FIG. 17. G. A. OF HYBRID SHEAR HINGE SYSTEM

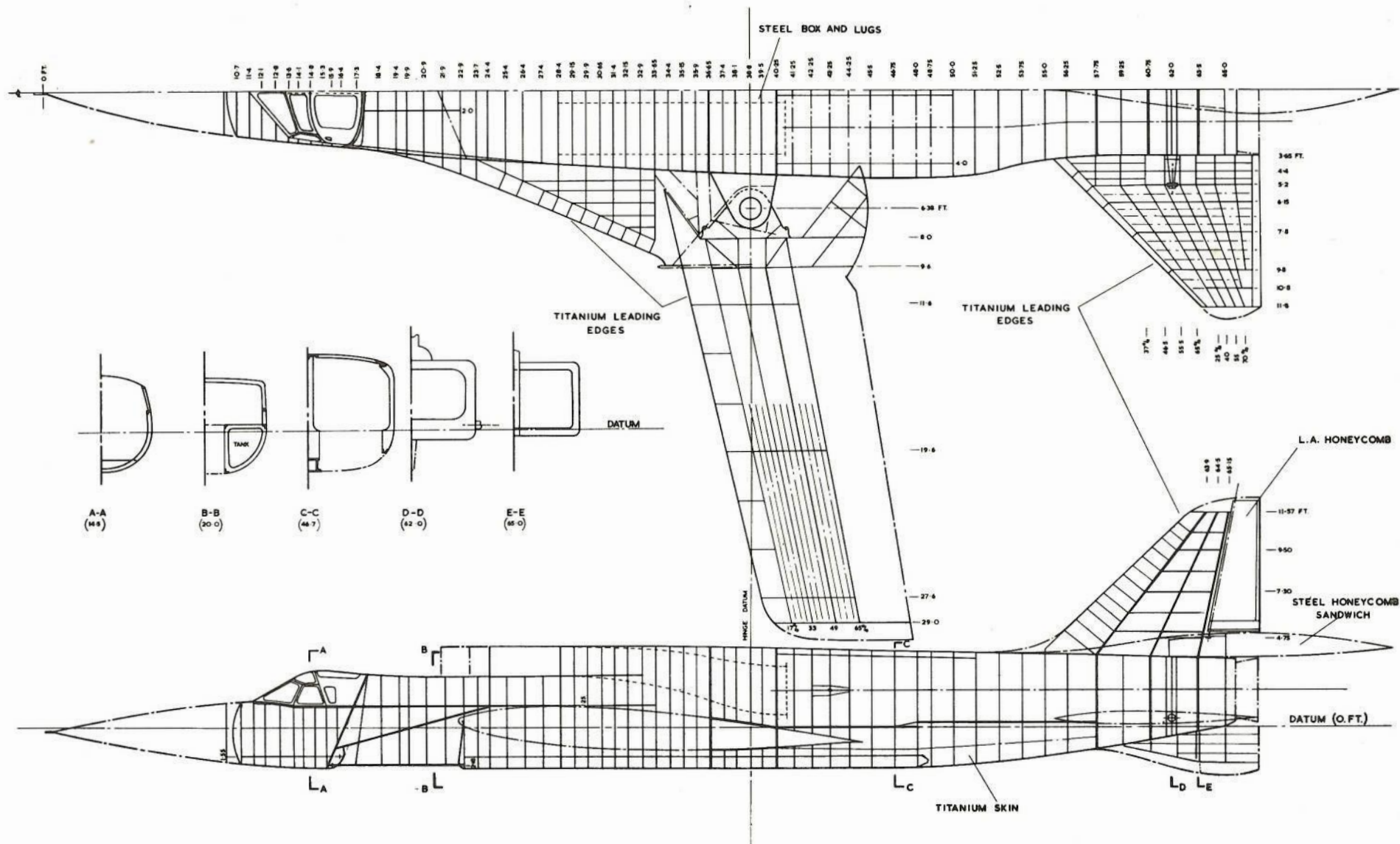


FIG. 18. STRUCTURAL LAYOUT

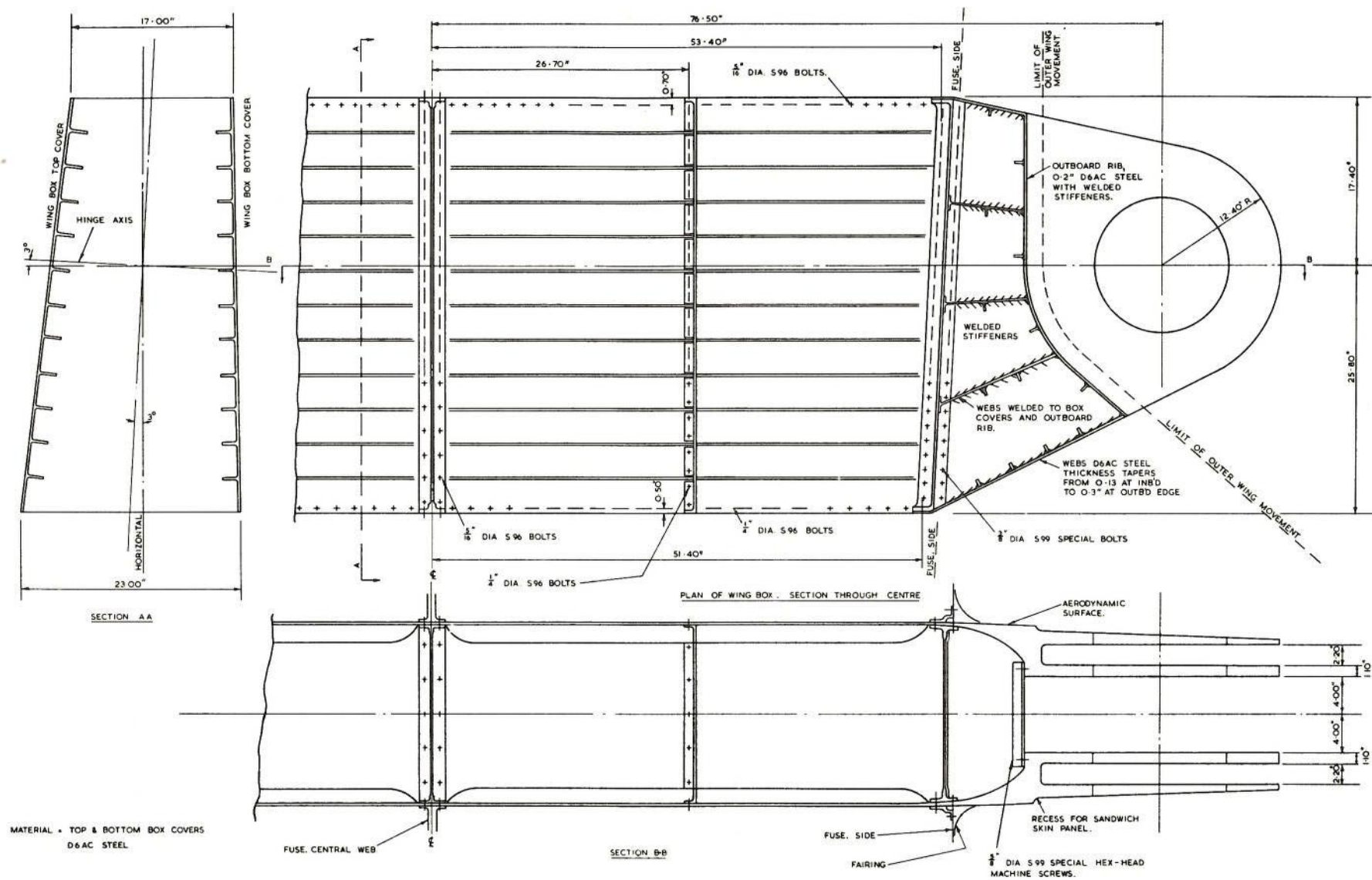


FIG. 19. HORIZONTAL SHEAR HINGE - CENTRE WING STRUCTURE

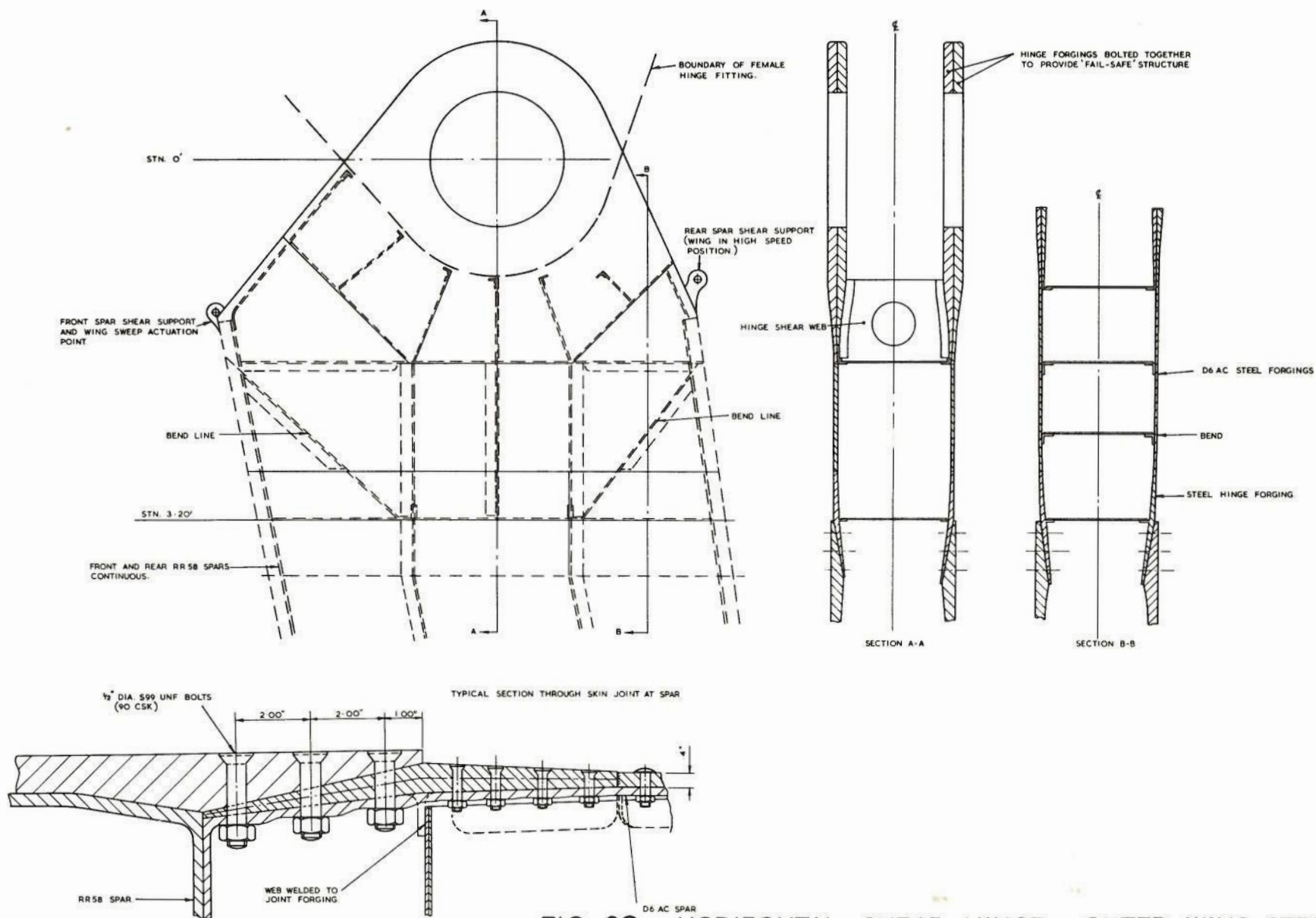


FIG. 20. HORIZONTAL SHEAR HINGE - OUTER WING STRUCTURE

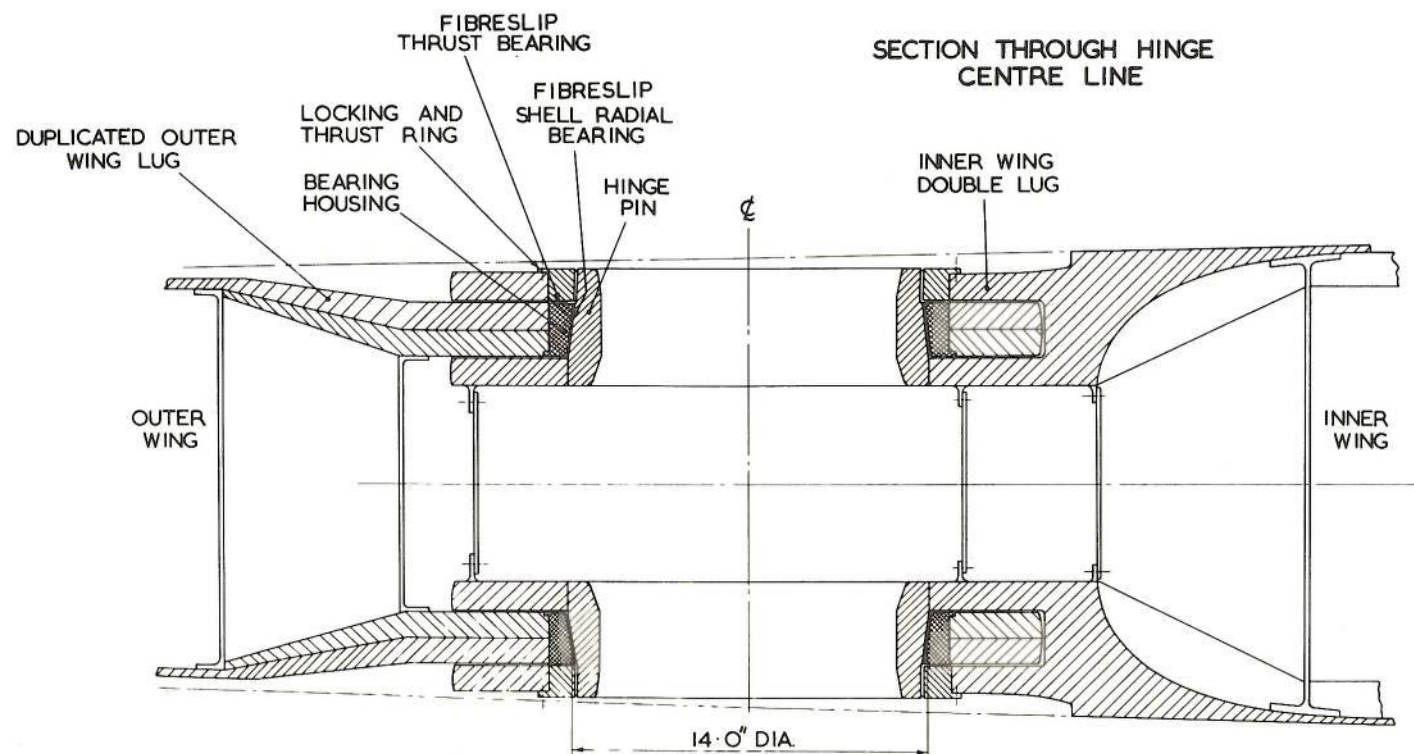


FIG. 21. SECTION THROUGH HORIZONTAL SHEAR HINGE

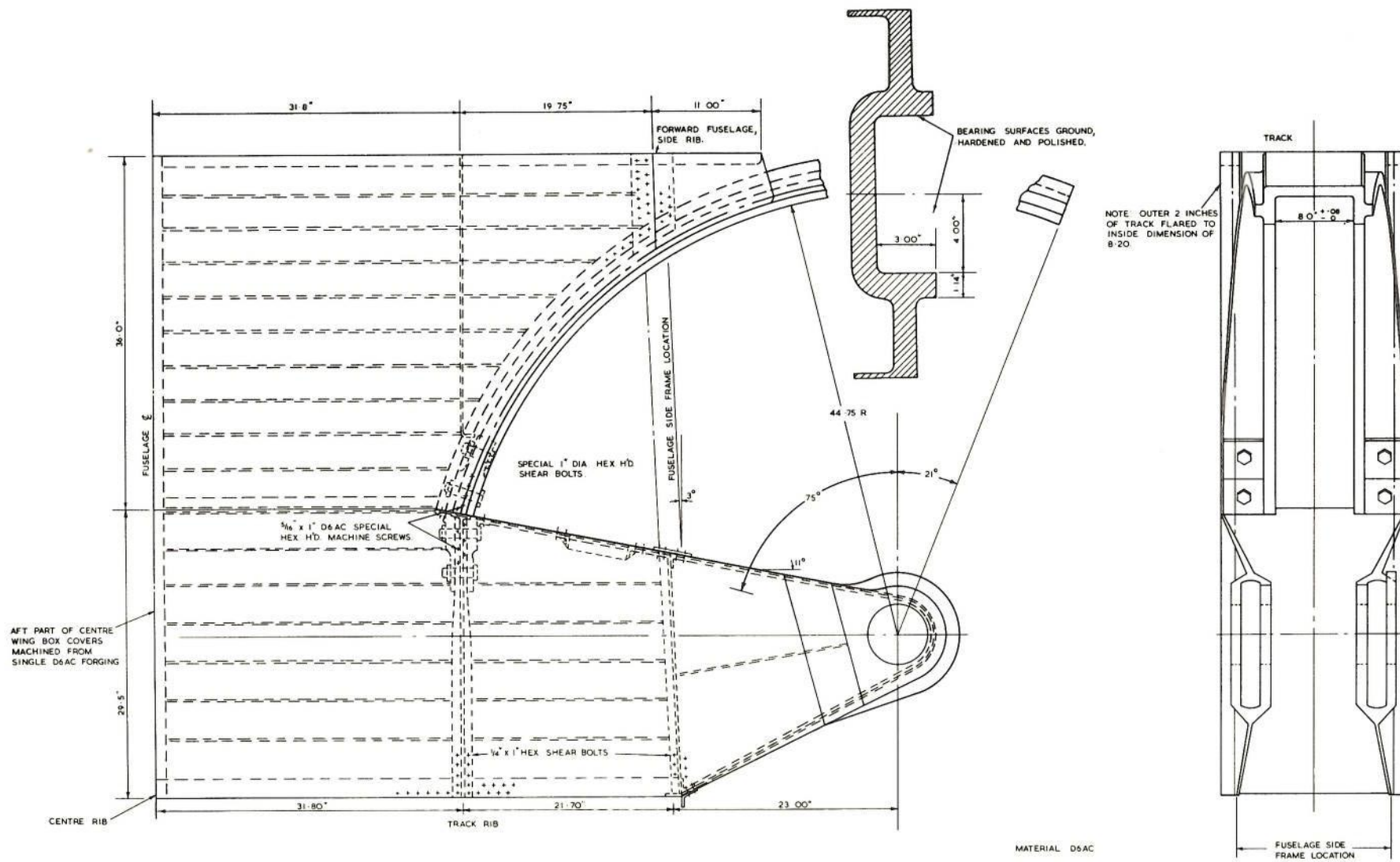
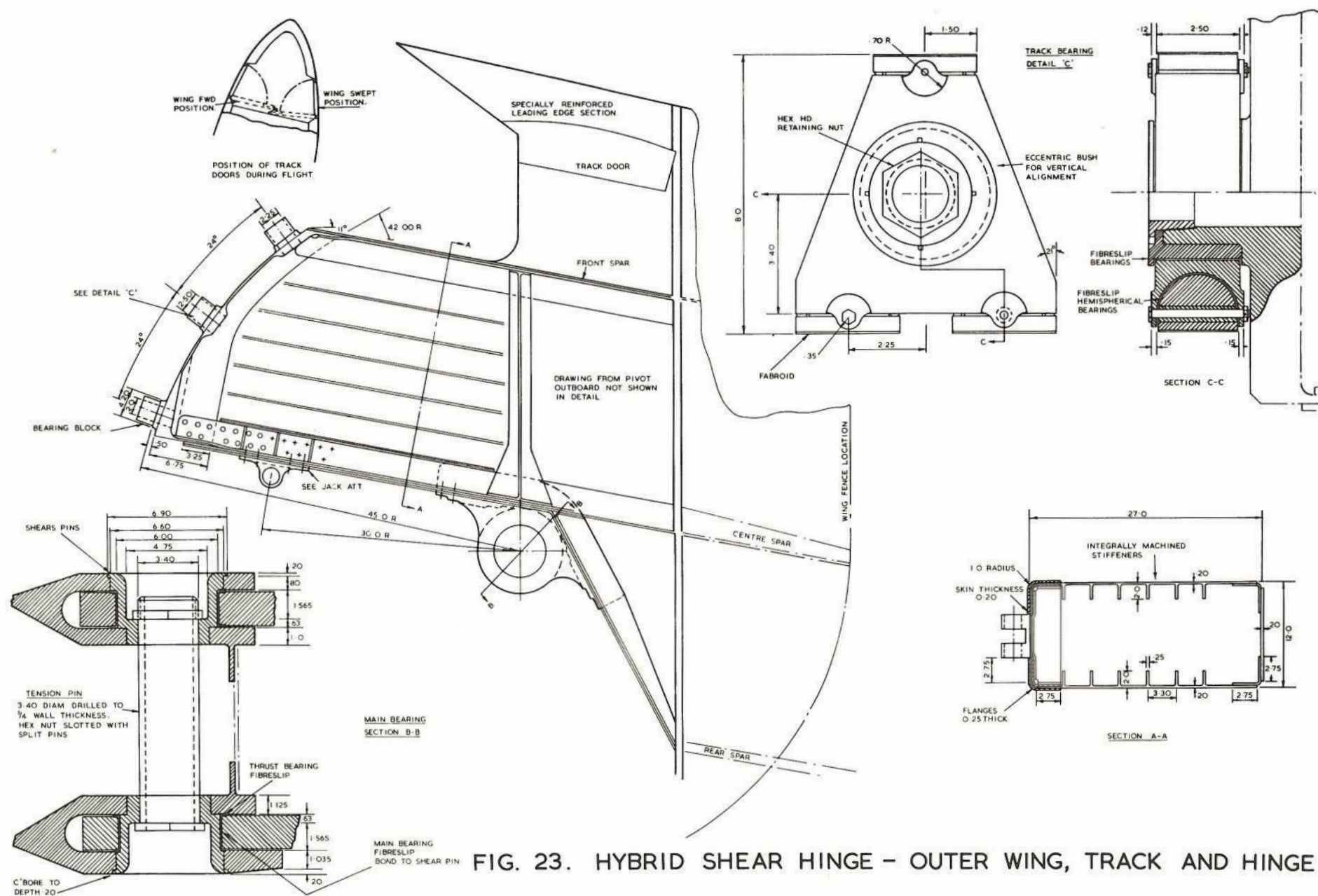


FIG. 22. HYBRID SHEAR HINGE - CENTRE WING STRUCTURE



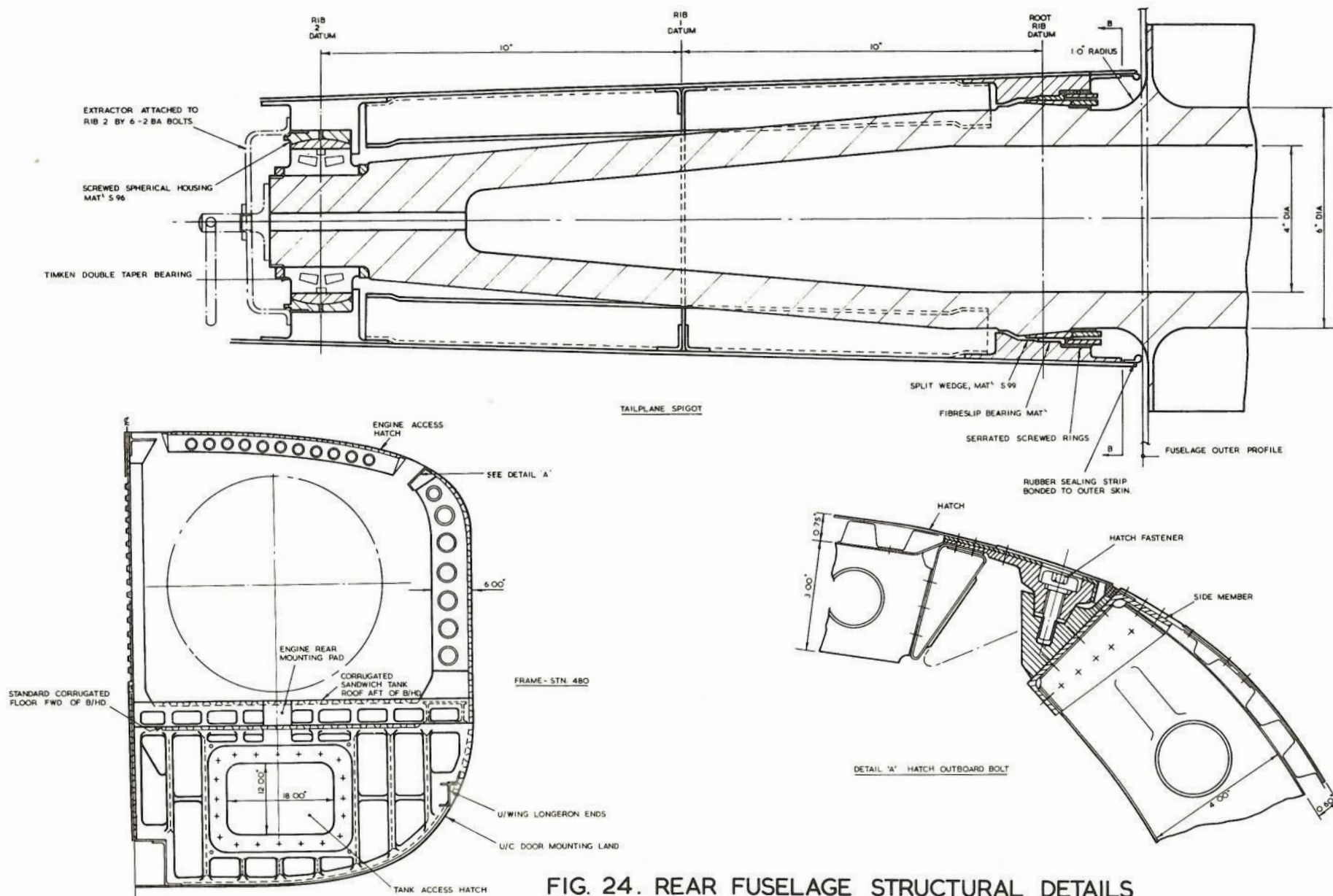


FIG. 24. REAR FUSELAGE STRUCTURAL DETAILS

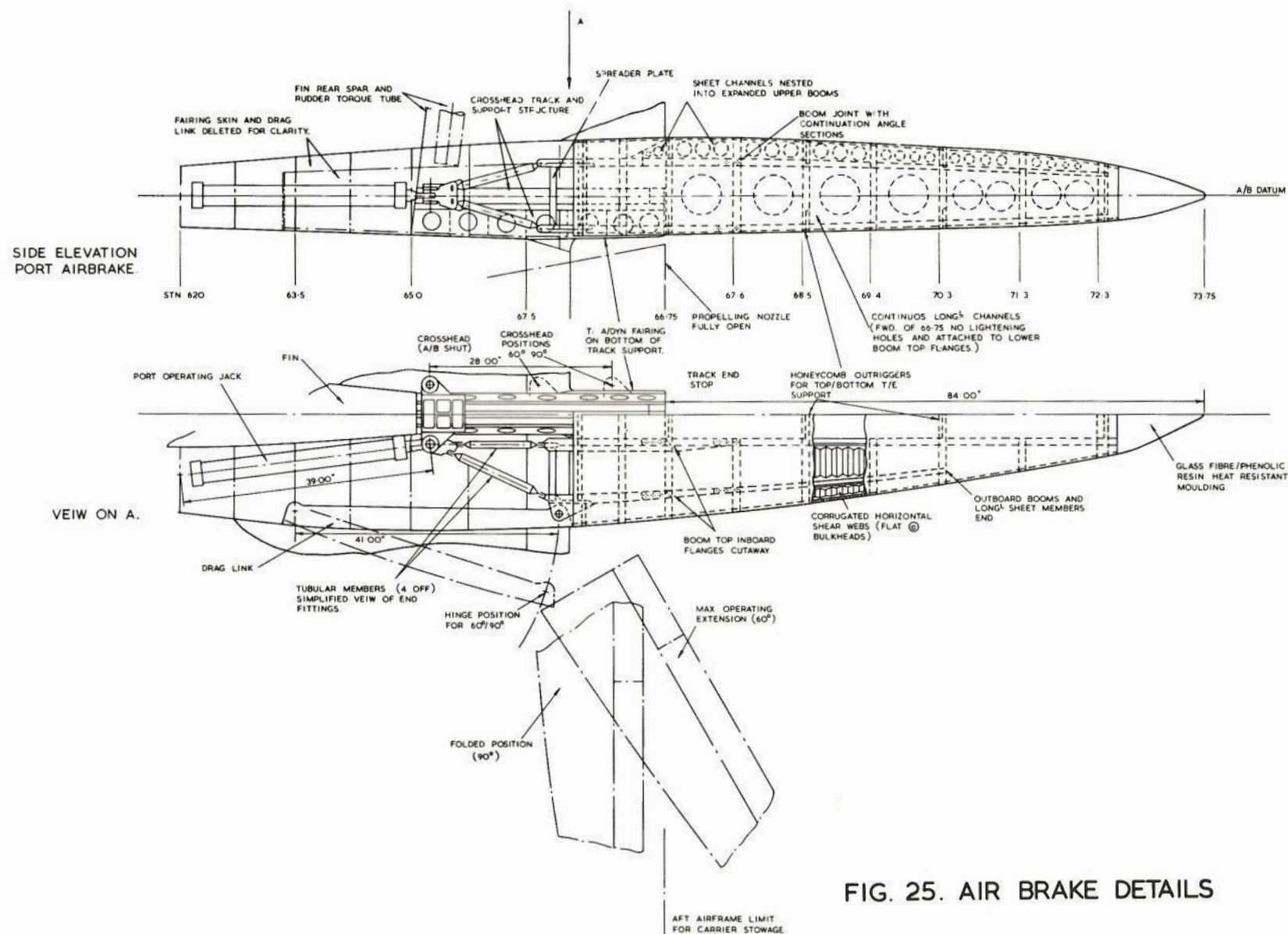


FIG. 25. AIR BRAKE DETAILS

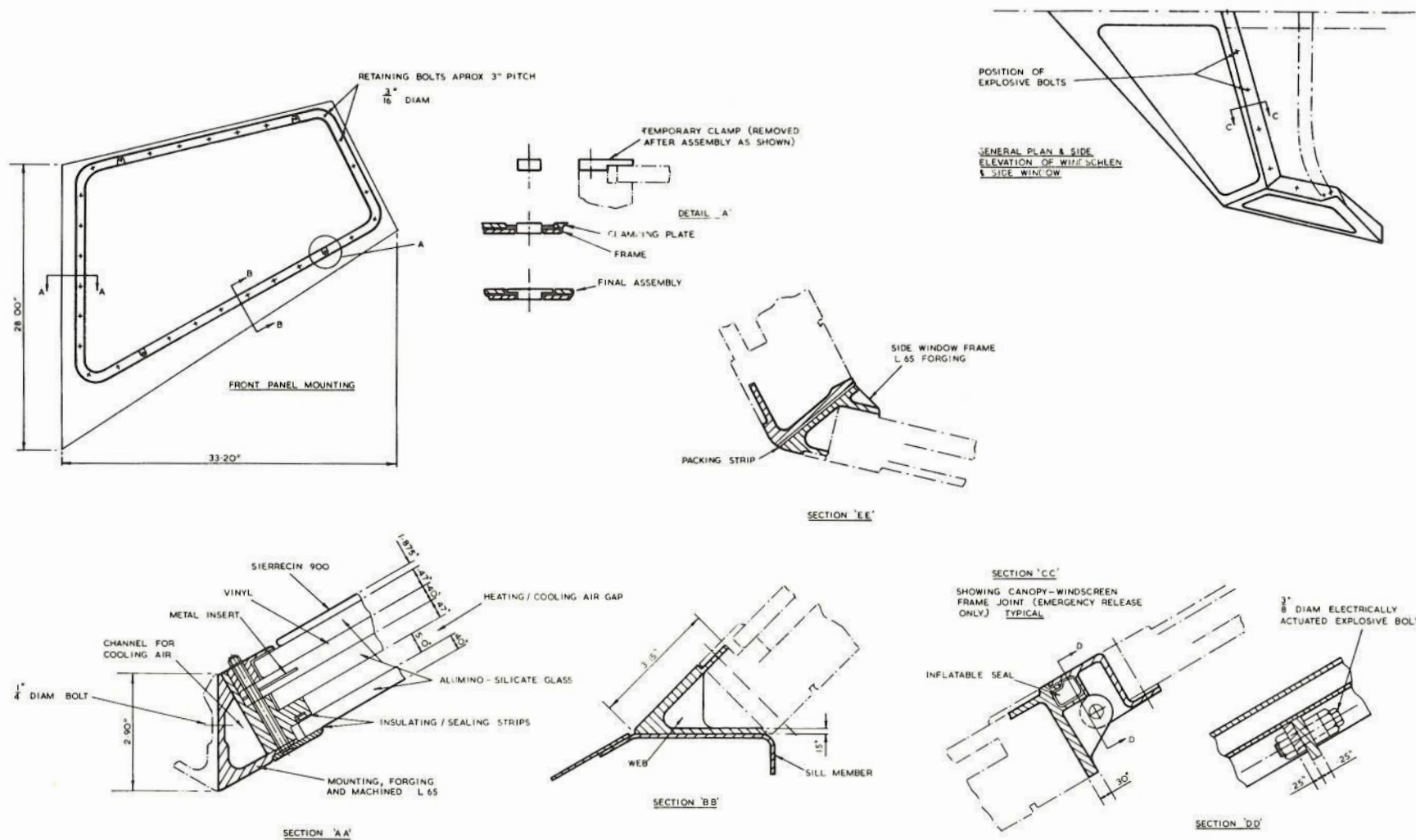


FIG. 26. WINDSCREEN AND CANOPY DETAILS

